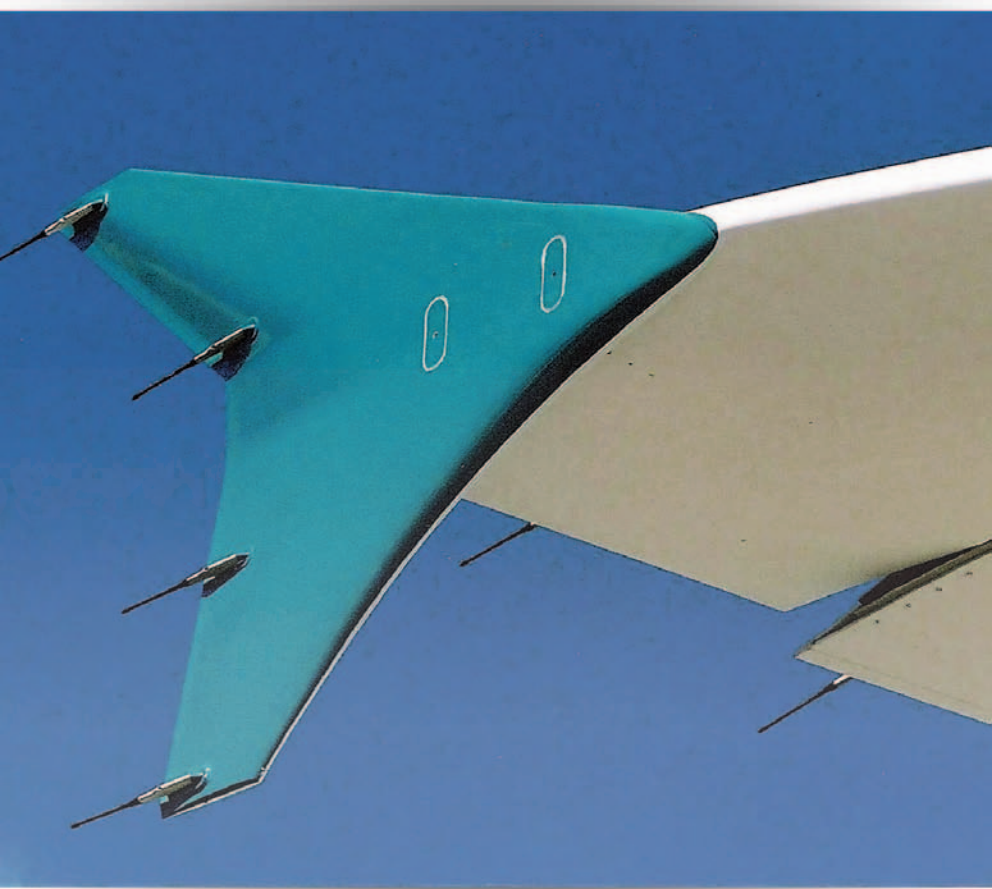


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# THE ENCYCLOPEDIA OF AERODYNAMICS

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FRANK E. HITCHENS

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# THE ENCYCLOPEDIA OF AERODYNAMICS

FRANK E. HITCHENS

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# Introduction

The Encyclopedia of Aerodynamics was written to cover all aspects of aerodynamics, which are of interest to pilots, aeronautical engineers, and students of aerodynamics. Aerodynamic references across the speed range are included from low-speed subsonic to supersonic and hypersonic aerodynamics, plus helicopter and propeller aerodynamics; entries for research aircraft, research facilities, test pilots, and aerodynamicists are also included, along with a cross-reference.

Aerodynamics is an applied science involving the motion of an aircraft moving through the air and the reaction of the air when disturbed by an aircraft. The name aerodynamics is taken from the Greek language where *aerios* (aero) means dealing with the air and *dynamis* means force. However, the subject can be further divided into classical (subsonic), transonic, supersonic and hypersonic aerodynamics. Propeller and helicopter aerodynamics is both an extension of classical aerodynamics and both have their own peculiarities and characteristics.

Fluid dynamics was the name the early scientists used when referring to what we now know as classical aerodynamics, which is based largely on Newton's three Laws of Motion taken from Newtonian mechanics. It deals with subsonic flow with velocities below Mach 0.8 and covers the concepts of air resistance, laminar and turbulent airflow, the boundary layer, Bernoulli's Theorem, separation and stagnation points, circulation, Reynolds numbers and vortices in an incompressible flow. Supersonic aerodynamics assumes the air is a compressible flow and introduces the terms of Mach number, shockwaves, aerodynamic heating and compressibility, etcetera. Transonic aerodynamics deals with the problems of flight between Mach 0.8 to Mach 1.2 encompassing

gradual changes in conditions between the subsonic and supersonic flight regimes.

When did all this information originate? The science of aerodynamics is a relatively new science as we know today; however some people maybe surprised to learn it first started many decades before aircraft first flew. It has its origins in the 16–17<sup>th</sup> Centuries when the early scientists were researching ballistics, hydraulics and fluid dynamics (the effect of water flow on ships, etc).

Who made these important discoveries leading up to the modern aerodynamics that we know today? Initially, we can thank Daniel Bernoulli (1700–1782) a contemporary of Isaac Newton and Bernoulli's colleague, Leonhard Euler (1707–1783) and Sir George Cayley (1773–1857) who some authorities consider to be the original father of aerodynamics for heavier than air flight. Many other great names have been involved with the development of aerodynamics, especially in the first half of the 20<sup>th</sup> Century. These names can be attributed to a select few – such names as Professors Adolph Büsemann, Nikolai Joukowski, Theodore von Karman, Martin Kutta, Ludwig Prandtl, Dr. Dietrich Küchemann and Richard Whitcomb. This list is by no means complete and several more names are mentioned within the pages of this book; however, my apologies to those not mentioned, who have also contributed greatly to the science of aerodynamics. Much of this early research originated in continental Europe – Switzerland, Germany, Russia, and also England and to a lesser degree in other countries. The big NACA/NASA research centres in the USA, which started in the 20<sup>th</sup> Century, did and still do contribute a great deal of aerodynamic research.

We should also remember the test pilots who flew the research aircraft, risking their lives (and sometimes losing it) to test and prove various aerodynamic theories. Pilots and aerodynamicists alike have all contributed a vast amount of information and knowledge to the science of aerodynamics to improve the safety of flight and of course, to make flight possible. With their valued input, the aviation world – and in fact the whole world – has

progressed in leaps and bounds to what we know today, with safe and efficient aircraft to transport people and freight from place to place around the world.

Aerodynamics is the fascinating science behind aircraft flight, a subject that all pilots (and aerospace engineers) must know at least a little about – and the more a pilot knows about aerodynamics, the safer he will be as a pilot. This book, *The Encyclopedia of Aerodynamics*, will answer many questions the reader has on this fascinating subject for the duration of his flying career from ab-initio student pilot to airline transport pilot, recreation pilot or military pilot, etcetera. Over sixteen hundred entries are included; many are cross-referenced to associated subjects and backed up with 97 diagrams and a selection of photographs.

The author's intention has been to present the subject of aerodynamics in an easy to read format and to keep mathematics to the very minimum, while focusing more on the physical aspects of aerodynamics. Several other books are available for the student covering engineering aerodynamics, involving higher mathematics and Calculus. All diagrams have been drawn free hand and are not to scale; therefore, they do not represent any particular aircraft. By convention, diagrams representing the airflow over an aircraft assume the aircraft to be moving from right to left; diagrams for propellers assume the propeller blade to be moving from left to right. It is also assumed the aircraft is stationary with the airflow moving over it, which is the opposite of reality. Keep this in mind to avoid any confusion on the airflow situation.

# Acknowledgements

All photographs are from the author's collection with the exception of the MD-80 Propfan test plane photograph, which was freely donated by Hamilton Standard of Connecticut, USA, to whom I am truly thankful.

Additionally, I would like to thank those people who have helped me with this book and arranging for me to use the photographs from my own collection of their aircraft. Namely, Sarah Swan from the National Museum Of the United States Air Force in Dayton, Ohio; Meghan Marum from the Pima Air & Space Museum in Tucson, AR; Barbara Gilbert from the Fleet Air Arm Museum in Yeovilton, UK; Karen Crick from the RAF Museum in Cosford, UK; Howard Heeley at the Newark Air Museum in Newark-on-Trent and Dianne James at the Midland Air Museum, Coventry, UK.

Finally, I dedicate this book to my beloved wife, Deirdre, who encouraged me during my early researching and writing this book. Unfortunately, she was not to see the final result.

Frank Hitchens,  
Wellington, New Zealand.

# A

**Absolute aerodynamic ceiling** The absolute aerodynamic ceiling is the maximum density altitude at which the aircraft's rate of climb has reduced to zero feet per minute, due to reduced engine power.

When an aircraft climbs to altitude, the thrust horsepower required increases and the thrust horsepower available decreases. Eventually an altitude is reached where these two requirements coincide at the absolute aerodynamic ceiling, there is no more excess thrust horsepower available for climbing, and thus, the rate of climb reduces to zero. The minimum and maximum level flight speeds along with the maximum angle and rate of climb speeds all converge down to one speed.

It is also known as the absolute ceiling, or aerodynamic ceiling.

*Compare with* **Combat ceiling; Service ceiling.**

**Absolute ceiling** *See* **Absolute aerodynamic ceiling.**

**Absolute density** The absolute density refers to the theoretical air density at a given altitude according to the International Standard Atmosphere (ISA) conditions. The sea level air density is assumed to be  $1.225 \text{ kg/m}^3$  or  $0.0765 \text{ lb/cu.ft.}$

*See* **Density.**

**Absolute pressure** The absolute pressure is equal to gage pressure plus the ambient atmospheric pressure. It is zero referenced against an absolute (perfect) vacuum.

Absolute pressure = gage pressure + ambient pressure.

The absolute pressure is used in **Boyle's Law** for all calculations.

**Absolute zero temperature** The absolute theoretical minimum temperature by international agreement at which all the molecular motion ceases is  $-273.15^{\circ}\text{C}$  or  $-459.67^{\circ}\text{F}$  or zero on the Kelvin and Rankine scales. The absolute zero temperature is found where a thermodynamic system has lost its energy, the entropy is at its minimum value and no more heat is available – it is at its absolute minimum.

Jacques Charles (1746–1823) a French physicist is credited with discovering circa 1787, the absolute zero temperature. Charles' Law states the volume of gas is directly proportional to its temperature. By plotting the volume of all gases versus temperature, Charles discovered the lines all converged at the same point on the graph indicating  $-459.67$  degrees Fahrenheit.

See **Temperature**.

**Accelerated stall** An accelerated stall is one entered from a speed well above the  $V_{S1}$  flaps up stall speed.

A rapid up-elevator movement, a steep angle of bank, or a vertical gust causing the aircraft to pitch up with some rate of climb possible can cause it. The aircraft stalls at a speed higher than the normal flaps-up stall speed entered by gradual up-elevator producing a zero rate of climb. This is proof the aircraft can stall at any air speed when the critical angle of attack is reached and exceeded.

The accelerated stall is also known as a high-speed stall, or g-break.

**Acceleration** An acceleration is the rate of change in direction, or velocity. Therefore, an increase in speed and/or a change in direction are considered acceleration. Acceleration is directly proportional to the unbalanced force and inversely proportional to its mass. Reducing speed (decelerating) is a negative acceleration.

Acceleration = change in velocity/time of velocity change.  
( $A = \Delta V / \Delta t$ ).



**Acceleration due to gravity** Acceleration due to gravity ( $g$ ) is the force of attraction acting vertically downwards due to the presence of the earth.

Briefly, acceleration is proportional to Force/Mass or alternately force is proportional to Mass times Acceleration. Moving a mass further from the centre of the Earth, renders a decrease in weight with mass remaining constant. Because the force due to gravity decreases with distance, the weight also decreases in proportion. The acceleration due to gravity ( $g$ ) also varies slightly at different positions on the Earth due to the Earth being an oblate spheroid shape. The earth is slightly flatter at the poles reducing its diameter to less than that at the equator. The acceleration at the poles is  $9.83 \text{ m/s}^2$  and at the equator, it is  $9.78 \text{ m/s}^2$ . At the poles, a body is closer to the centre of the earth and so its acceleration is greater. The internationally accepted average value is  $9.8066 \text{ m/s}^2$  or  $9.8 \text{ m/s}^2$  measured at sea level at Latitude  $45^\circ$  north, which is the value used for practical purposes, sometimes rounded up to  $10 \text{ m/s}^2$ . The acceleration due to gravity is more commonly known as the 'acceleration of free fall'. Note the correct term is acceleration due to gravity, not the force of gravity.

Galileo Galilee (1564–1642) was an Italian astronomer, physicist and one of modern science's founding fathers. He put forward a hypothesis now named after him, which states 'at a given place on earth all objects fall with the same acceleration of  $9.8 \text{ m/s}^2$  ( $32.2 \text{ ft/s}^2$ ) if they are not affected by air resistance.' A feather and a lump of lead are usually quoted as examples; in a vacuum, they both fall at the same rate due to a lack of air resistance. However, due to greater air resistance in ambient air a feather will fall slower than a lump of lead. Given sufficient height, all falling bodies will reach a certain speed known as their terminal velocity, where they will no longer accelerate. The air resistance will equal the acceleration due to gravity; that is why the feather will fall slower.

Gravity is the physical attraction exerting between the Earth and another body (for example, airplane). In 1687, Sir Isaac Newton stated that 'the force of gravity is proportional to the product of

the mass of two bodies and inversely proportional to the square of the distance between them'. However, it was an English scientist, Henry Cavendish (1731–1810) who discovered the value of acceleration due to gravity  $9.8 \text{ m/s}^2$  ( $32.2 \text{ ft/s}^2$ ) towards the end of the 18<sup>th</sup> Century.

The acceleration due to gravity is also known as the Law of Gravitation or Newton's 4<sup>th</sup> Law.

**Acceleration in a turn** Acceleration in a turn is measured in  $\text{m/s}^2$  or  $\text{ft/sec}^2$ . These are the same units as used for an aircraft linear acceleration in straight and level flight when it experiences an acceleration of 1 'g'. If the aircraft now enters, a turn with a  $60^\circ$  angle of bank an additional stress force acceleration of 1 g will be applied. The aircraft (and its occupants) will then experience an acceleration of 2 g's, imposed on the airplane, known as wing loading. The inertia force required to provide the acceleration towards the centre of the turn is expressed in terms of gravitational acceleration measured in 'g'. An aircraft in a constant rate turn at a constant air speed is considered to be accelerating because its direction is changing.

Note: a capital 'G' or lower case g, represents a stress force, and a lower case 'g' (with quote marks) represents linear acceleration, or acceleration due to gravity.

Acceleration towards the centre of the turn is given by:

$$V^2/R \text{ meters per second.}$$

The required centripetal force to produce the turn is given by:

$$\text{Force} = W V^2/gr \text{ Newtons.}$$

The correct angle of bank for the aircraft's speed is independent of the weight.

$$\text{Angle of bank } \tan^\circ = (W V^2/gr) \div W = V^2/gr.$$

**See Turning; Centripetal Force; Centrifugal Force; Radius of turn; Rate of turn.**

**Ackeret's theory** Jacob Ackeret (1898–1981) suggested in 1925 that an aircraft's wing in supersonic flow should have a low thickness/chord ratio and a sharp leading and trailing edge. His theory has proven to be correct.

**Ackeret, J.** The Swiss national, Jacob Ackeret (1898–1981) is noted for being the first to study supersonic flow past an infinite wing using a supersonic wind tunnel, which he designed and built. An infinite wing can be considered as a wing in a wind tunnel stretching from wall to wall where the effects from the wingtip vortices are ignored. Through his discoveries in 1925, he stated a wing operating in a supersonic flow should be of low thickness/chord ratio (a thin wing) and have sharp leading and trailing edges, a profile now adopted for all high-speed airplanes. He also discovered the drag diversion phenomena.

At the fifth Volta Conference in Rome, Italy on the 30 September 1935, Jacob Ackeret gave a talk on the supersonic wind tunnels he had designed, which were built in Switzerland, Italy, and Germany around that time. Jacob Ackeret also coined the term Mach number in honor of Ernst Mach (1838–1916).

**Active aeroelastic wing** Future jet fighter aircraft may benefit from the active aeroelastic wing concept to enhance their combat manoeuvrability by using the wing's inherent aeroelastic properties to advantage.

*See Mission adaptive wing.*

**Activity factor** The activity factor of a propeller is a measure of the quantity of power a propeller can absorb. Each different type of aircraft can accept a range of engines of varying amounts of brake horsepower, within certain limits. Likewise, each engine can accept a variety of propellers, also within certain limits. One of these limitations is the propeller's ability to absorb the power provided by the engine. For most piston-engine aircraft, the

maximum thrust horsepower available is approximately 85% of the engine's brake horsepower. If the propeller is absorbing 83% of the engine's total brake horsepower output, the activity factor is quoted as 0.83. The activity factor is just another way of expressing the propeller's efficiency.

**Actuating wingtip** The actuating wingtip is an advanced form of winglet for supersonic aircraft. A normal winglet is a fixed device but the actuating wingtip is drooped down at high-speeds to help provide compression lift. The only aircraft equipped so far is the North American Aviation XB-70 Valkyrie Mach 3.0, supersonic bomber/research aircraft with actuating wingtips drooping to 60°, and the British BAC TSR-2. Neither aircraft went into production.

*See* **Compression lift; Research aircraft — North American Aviation XB-70A Valkyrie.**

**Actuator disc** The actuator disc was introduced by William Froude (1810–1879) when he introduced his theoretical propeller disc, which is associated with the axial momentum theory (or disc actuator theory) of propeller propulsion.

*See* **Froude, W.**

**Adiabatic process** This is a thermodynamic process where there is no heat transfer either in or out of the system. When a gas (air) is compressed, the temperature rises; it cools with expansion. Expenditure of energy is a requirement to raise the temperature of the gas but due to the conservation of energy, the energy in the gas is not lost but transformed into molecular energy motion. A common explanation of adiabatic heating is the bicycle pump, which heats up when used to inflate a tire. The friction drag of fast flying aircraft also produces heat.

*See* **Thermodynamics; Isentropic flow; Isobaric; Open cycle; Entropy; Isothermal.**

**Advance/diameter ratio** The advance/diameter ratio is a parameter that can be used to define the characteristics of a propeller using a non-dimensional form. The advance/diameter ratio is the ratio of the airplane's velocity (true air speed) to the product of propeller RPM and diameter. The advance/diameter ratio is:

$$J = V/nd:$$

Where J = advance diameter ratio

V = true air speed

n = RPM in unit time

d = prop diameter.

See **Effective pitch ratio; Slip function.**

**Advance per rev** The advance per rev (APR) is the distance the propeller advances forward in one revolution.

The advance per rev is also known as the induced inflow velocity.

See **Induced inflow & angle.**

**Advanced stall** An advanced wing stall is one that is past the incipient stage and well developed to become a fully developed stall, possibly with a wing dropping as lateral stability is compromised.

See **Stall break; Stalling angle; Stall progression.**

**Advancing and retreating blades** A helicopter in forward flight has its advancing rotor blade(s) moving in the same direction of flight, while the retreating blade(s) are moving in the opposite direction. The advancing blade will be moving forward at the speed of the helicopter plus its own forward speed due to rotation. The opposite is true for the retreating blade – that is, the helicopter's forward speed minus the speed of the retreating rotor blade.

On a helicopter, moving sideways or backwards, the advancing blade (and retreating blade) are relative to the direction of helicopter movement.



**Photo:** The NH 90 helicopter in forward flight. The advancing blade is on the right side of the helicopter and the retreating blade on the left.

**Advancing blade concept** Sikorsky Aircraft developed the Sikorsky XH-59A-ABC helicopter to research the advancing blade concept (ABC).

The rotor blades on a conventional helicopter are restricted to a maximum speed due to retreating blade stall. However, the advancing blade concept on the Sikorsky XH-59A-ABC uses two, 36 foot diameter rigid contra-rotating rotor discs one above the other. Therefore, there is always a simultaneous advancing blade on each disc to provide the necessary lift for flight and of course, a retreating blade on each disc. Dissymmetry of lift is eliminated. The advancing blades produce high dynamic pressure for lift as opposed to the retreating blades, which virtually do no work at all – they are unloaded. The advancing blade concept allows a much faster forward speed than a conventional rotor system, using airfoils optimized for high-speed performance. Due to the stiff rotor blades, the helicopter is very manoeuvrable making it ideal as a military combat helicopter.

However, the Sikorsky XH-59A never went into active service due to the weight disadvantage of the rotor head far outweighing its fast speed and manoeuvrability. The XH-59A entered the joint NASA/Army test program in 1971 with two models built for testing. As a pure helicopter powered by two PT6T-3 Turbo Twin-Pac engines, it achieved a top speed of 184 MPH in level flight and 221 MPH in a dive. With two additional P&W J60 fuselage-mounted jet engines for increased forward thrust, the top speed was approximately 300 MPH. In this configuration, the rotor blades were virtually windmilling in autorotation at a lower RPM than normal. The program ended in 1977 with the loss of one craft due to a heavy landing.

**Adverse aileron yaw** Application of aileron to roll the aircraft into a turn in a given direction can result in the aircraft yawing away from the required turn, due to the effect of adverse aileron yaw. The effect is less pronounced on modern types of aircraft built since circa WW II, which are equipped with Frise or differential ailerons. However, it can manifest itself, particularly during flight at relatively low-speeds due to the greater aileron deflection required to produce a given rate of roll.

The initial application of aileron is to cause the aircraft to roll in the required direction of turn. On the rising wing, the relative airflow approaches the wing from an angle above that of the flight path and the lift vector is therefore, inclined rearwards, and produces a retarding force against the required direction of turn. Added to this, the increase in angle of attack of the rising wing causes extra aileron drag. The rising wing yaws away from the required direction of turn, instead of towards it.

The converse is true on the descending wing, although it still aids the adverse yaw effect. The relative airflow approaches the descending wing from below causing the aerodynamic lift vector to be tilted forwards; this vector pulls the wing forwards and away from the required direction of turn. The drag of the aileron is also reduced due to the reduced angle of attack. The result is the combination of tilted lift vectors causes a yawing moment and the difference in drag from both ailerons causes adverse aileron yaw. Application of rudder is usually required to coax the aircraft to turn in the required direction. Frise and differential ailerons largely alleviate the problem of adverse aileron yaw.

Aircraft with a high aspect ratio wing, short fuselage, or small area tail fin are more prone to adverse aileron yaw.

*See* **Frise ailerons; Differential ailerons.**

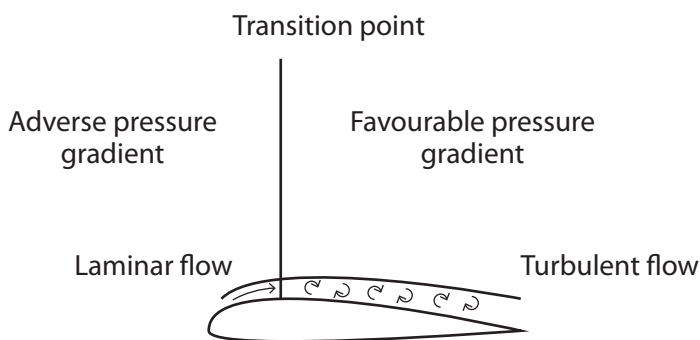


**Adverse pressure gradient** When a fluid element (air particle) flows over the upper surface of the wing its velocity reduces, due to increasing surface friction. This is a disadvantage to the airflow and is therefore termed an adverse pressure gradient (increasing pressure).

Due to the increasing pressure gradient and skin friction, the velocity is further retarded and may eventually become stationary or even reverse direction (recirculation) in the turbulent flow. The turbulent flow causes separation of the boundary layer at the separation point, a loss of lift, an increase in drag and eventually the wing stalls at the critical angle of attack. The air pressure at the wing's trailing edge determines the severity of the adverse pressure gradient.

The adverse pressure gradient is also known as the reverse pressure gradient, or the trailing-edge condition, where the air pressure changes from low to high pressure, as it flows over the wing.

*See also* **Boundary layer; Separation point & flow.**



**Diagram 1, Adverse and Favourable Pressure Gradients**

**Aerobatic category** All aircraft certified in the aerobatic categories have no restriction on flight manoeuvres up to the limit load factor for that type, which must be a limit load factor of at least positive 6.0g. The aerobatic category also includes all manoeuvres permitted in the normal and utility categories.

*See* **Normal category; Limit & ultimate load factors; Ultimate load factor.**



**Photo:** An MXT is a spectacular aerobatic performer.

**Aerodynamic axes** An aircraft in flight moves in three dimensions around its centre of gravity (O origin). Its attitude is described with reference to the three aerodynamic axes, which are all perpendicular to each other and follow the right-hand rule.

The three axes are the OY lateral, OX longitudinal and OZ vertical axis, about which the aircraft pitches, rolls and yaws respectively. **Diagram 2, Planes of Movement**, shows a Cartesian coordinate system representing the aircraft's three axes.

The OX longitudinal axis acts in a forward direction in line with the aircraft's centre line. The aircraft's movement about the OX longitudinal axis is considered positive when the aircraft rolls to the right. The OZ vertical axis acts vertically downwards with yaw being positive when the aircraft yaws to the right. The vertical axis is also known as the ground axis or gravity axis. The OY lateral axis is considered positive when the aircraft pitches nose-up around this axis. The three axes collectively are also known as the body axes or principle axes of inertia.

The following table lists the three axes, direction of positive movement, primary and secondary effects of movement. Note, axes are the plural and axis is the singular.

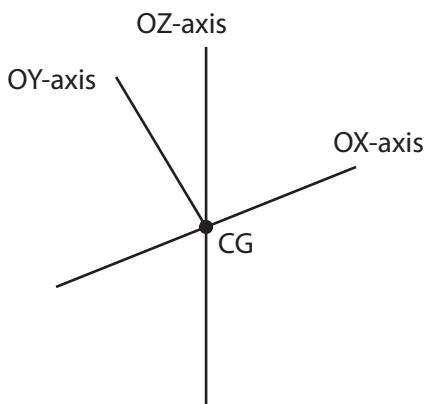
| Axis            | Direction  | Primary effect | Secondary effect |
|-----------------|------------|----------------|------------------|
| OX Longitudinal | Roll right | Roll           | Yaw              |
| OY Lateral      | Pitch up   | Pitch          | Air speed        |
| OZ Vertical     | Yaw right  | Yaw            | Roll             |

**Aerodynamic balance** Aerodynamic balance is used to reduce the hinge moment on the primary flight control surfaces, namely, the rudder, ailerons and elevator. The aerodynamic balance helps to balance the aerodynamic force acting on the control surface; thus, it relieves part of the stick force required by the pilot to move the control surface and thus improves control harmony.

The distance of the control surface's centre of pressure from its hinge line determines the stick force required to move that control. If the hinge moment is too great, the control will be heavy to move. Aerodynamic balance, and harmony, is achieved by careful design and the use of balance tabs, horn balance, or inset hinges, which are all various methods of achieving the same result.

Note mass balancing is used for an entirely different purpose and that is to prevent control flutter.

*See* **Horn balance; Inset hinge; Balance tab; Anti-balance tab.**



**Diagram 2, Planes of Movement**

**Aerodynamic ceiling** *See Absolute aerodynamic ceiling.*

**Aerodynamic centre (a. c.)** The aerodynamic centre is the point on a cambered airfoil about which the pitching moment is zero with changing angle of attack.

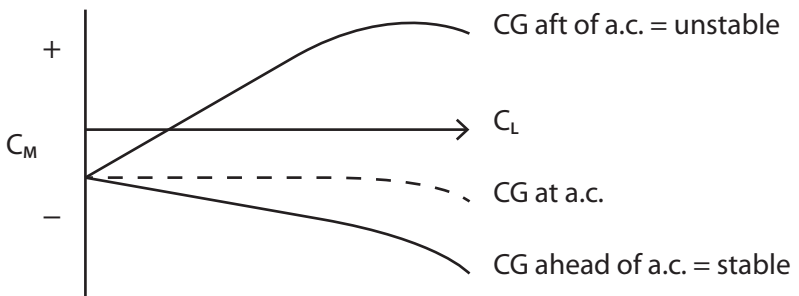
Due to the pressure distribution circulating around the airfoil, the forces acting on the airfoil's surface will vary with changes in angles of attack. On a theoretical, low-speed cambered airfoil section in an incompressible flow, the aerodynamic centre is located at 25% mean aerodynamic chord (MAC), and changes in lift, angle of attack, camber, or thickness have no effect on this location. If the wing experiences compressibility, the aerodynamic centre may be slightly forward or aft of 25% MAC (a plus or minus 2% variation) and move even further rearwards to about 50% MAC on aircraft at supersonic speed. Changes in aerodynamic stability and trim are induced when subsonic aircraft experience high Mach numbers for which they are not designed. On a symmetrical airfoil, the aerodynamic centre lies upon the chord line, or slightly above or below it, in the same location as the centre of pressure. There are no pitch change moments about the aerodynamic centre, the centre of pressure and aerodynamic centre being co-located. However, on a cambered airfoil, the centre of pressure moves fore and aft with changes in angle of attack, and hence changes in the lift distribution on the wing. Supersonic aircraft require relatively large elevators for manoeuvring in supersonic flight, due to the increase in static longitudinal stability caused by the aft shift in the aerodynamic centre location.

With changing angle of attack, the pitch-down moment increases at the wing's leading edge and decreases at the trailing edge. At the aerodynamic centre, the pitching moments balance each other and therefore are zero with changing lift coefficient.

Aerodynamicists find it more advantageous in their calculations for the whole aircraft to use the aerodynamic centre of the wing, which remains stationary, rather than the centre of pressure that moves fore and aft with changes in the angle of attack. Dr. Robert Rowe Gilruth (1913–2000) introduced the term aerodynamic centre.

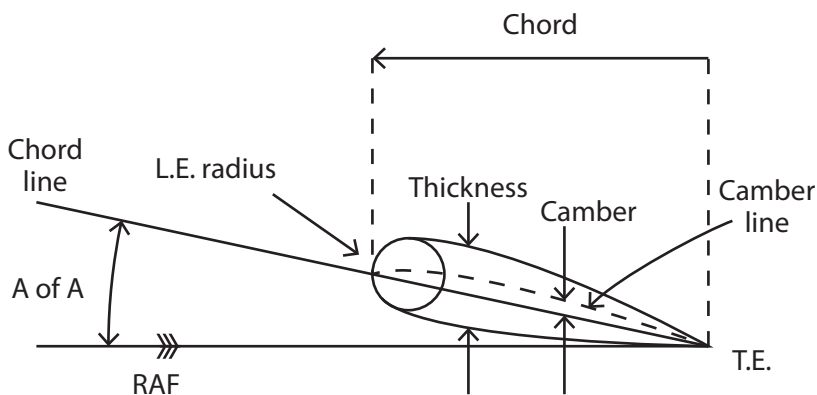
It is also known as the axis of constant moments.

*Compare with* **Centre of pressure; Centre of dynamic lift.**



**Diagram 3, Aerodynamic Centre & Pitching Moments.**

**Aerodynamic chord** The wing's chord is taken to be the distance between the leading and trailing edges measured at the wing root (root chord) unless quoted otherwise. The wingtip chord (tip chord) is also used for measurements, where the ratio of the tip chord/root chord is equal to the taper ratio. The average chord or geometric chord is the distance halfway between the root and tip chords and is used for example, in aspect ratio calculation where the span is divided by the chord ( $b/c$ ).



**Diagram 4, Airfoil Terminology**

The chord may be increased by leading edge devices, which extend forward relative to the leading edge and at the trailing edge by area increasing flaps such as Fowler flaps.

See **Aerodynamic chord; Aerodynamic root reference chord; Mean aerodynamic chord (MAC).**

**Aerodynamic coefficients** The aerodynamic coefficients are frequently found in aerodynamic formulas, e.g., lift coefficient ( $C_L$ ), drag coefficient ( $C_D$ ) and moment coefficient ( $C_m$ ) etc. For example, in the lift formula,  $\text{Lift} = C_L \frac{1}{2} \rho V^2 S$ .

See **Drag coefficient; Lift coefficient; Moment coefficient.**

**Aerodynamic coupling** Aerodynamic coupling is due to low directional stability. It involves the combined aerodynamic effect of yaw and roll axes where movement of one axis affects movement in the other.

*See Inertia coupling.*

**Aerodynamic curve** *See Wing load-grading curve.*

**Aerodynamic damping** Aerodynamic damping is a restoring moment, which restores an airplane back to balanced flight following a disturbance. An oscillating motion is damped out by the dissipation of energy due to the aerodynamic damping moment. The oscillations associated to longitudinal dynamic stability for example, are required to be damped out (as are the oscillations in the lateral and yaw planes).

Assuming a wing drops due to turbulence, the airplane will briefly sideslip towards the lower wing. The relative airflow will then have a vertical component from below causing an increase in the angle of attack and increased lift on the descending wing. The opposite effect occurs on the rising wing where the vertical component of the relative airflow descends from above reducing the angle of attack and lift. The combined airflow effect on both wings results in a restoring moment to balanced, wings level flight. Jet transports typically have a yaw damper installed to damp out any yawing tendency during flight.

Negative damping also exists in the form of control flutter, for example.

**Aerodynamic decalage** *See Decalage.*



**Aerodynamic drag** Aerodynamic drag is the resistance due to the air flowing over the surface of the aircraft. The drag is subdivided into separate groups depending on the cause of drag.

The aerodynamic drag of a propeller blade or helicopter blade is shown on *Diagram 68, Propeller Pitch*, as the vector labeled propeller drag. The blade's aerodynamic drag acts in line with, and downstream of the relative airflow vector (vector A-C).

Propeller or rotor blade aerodynamic drag should not be confused with propeller or rotor blade torque, which is also shown on *Diagram 41, Hovering Forces*.

See **Drag**.

**Aerodynamic efficiency** An aircraft's aerodynamic efficiency is measured by its lift/drag ratio.

*Diagram 55, Lift/drag Ratio, Vectors & Graph*, shows the maximum lift/drag ratio is achieved at  $3^{\circ}$ – $4^{\circ}$  angle of attack. Above and below this figure the lift/drag ratio and hence efficiency is reduced. In normal flight attitudes, the aerodynamic resultant force (lift) is perpendicular, or nearly so to the flight path; the angle of attack is then at its most efficient angle. Lift is generally a constant force for most speeds in straight and level flight, as opposed to the drag force that varies considerably with changes in air speed. The maximum aerodynamic efficiency is therefore found at the maximum lift/drag ratio.

See **Propeller efficiency**.

**Aerodynamic force coefficient** The force coefficient is a dimensionless ratio between average dynamic pressure (aerodynamic force per unit area) and air stream dynamic pressure.

The aerodynamic force coefficient associated with lift and drag is determined by the following factors:

- angle of attack
- profile shape
- air density
- aircraft velocity
- wing area
- viscous flow effects
- compressibility effects

The last two factors of viscous flow and compressibility can be ignored for incompressible flow. To highlight the importance of the force coefficient  $C_F$  the formula can be shown as:

$$C_F = \frac{\text{force}}{qS}$$

Where force = aerodynamic force

$C_F$  = aerodynamic force coefficient

$q = \frac{1}{2} \rho V^2$

$S$  = wing area

See **Section lift coefficient ( $C_l$ )**.

**Aerodynamic forces** Aerodynamic forces are influenced by fluid viscosity and compressibility, which is addressed by the Reynolds number and Mach terminology, respectively.

Sir George Cayley (1773–1857) defined the aerodynamic forces of lift, drag, thrust and weight and their relationship with each other. The aerodynamic forces of lift and drag are due to the action of air pressure and shear stress on the aircraft. The various forces are all related and describe the effects of the aerodynamic reaction and weight on an aircraft or helicopter, etc.

See **Viscosity; Compression flow.**

**Aerodynamic heating** Aerodynamic heating is due to the skin friction and adiabatic compression of the boundary layer airflow over the aircraft's surface.

Air friction decreases the speed of the airflow over the aircraft's surface thus decreasing the kinetic energy in the airflow, which manifests itself as internal energy causing aerodynamic heating of the surface. The greater the speed the greater is the surface friction, which increases the surface temperature as more kinetic energy is dissipated. The greatest temperature rise over the aircraft occurs at its stagnation points, being a combination of stagnation temperature rise, ram temperature rise, plus ambient temperature.

The temperature rise below Mach 1.0 is relatively insignificant, however above Mach 1.0, the temperature rises exponentially. The altitude of flight is less important than the aircraft's speed except at lower altitude where the ambient temperature is warmer and combined with the ram temperature rise the temperature acting on the aircraft is increased. In hypersonic flight, the aerothermodynamic phenomena of increased temperature in a blunt body, bow shock wave is proportional the square of the Mach number.

Aerothermodynamic heating applies to air speeds above Mach 5.0. In aerothermodynamics, the variation in the temperature and composition of the gas at hypersonic speeds due to



**Photo:** The Bristol 188 research aircraft was built of stainless steel to research high-speed aerodynamic heating.

thermal and dynamic phenomena, must be taken into account. Oxidation, dissociation and ionization (known collectively as aerothermochemical processes) common to hypersonic flight changes the chemical constituents and electrical conductivity of the air (gas) flowing over the craft. Dissociation occurs around Mach 7.0 and ionization occurs beyond Mach 12.0: this is an inherent problem for re-entry vehicles.

The aircraft's surface temperature increases approximately in proportion to the speed squared ( $V^2$ ). During supersonic flight, the intense shockwave forming around the aircraft's nose results in an enormous increase in temperature rise.

The table below shows the approximate increase in temperature with increasing air speed, with speed given in Knots. To the temperature rise, add the static air temperature to find the actual temperature of the aircraft's surface. The approximate temperature rise is calculated from the following formula, where speed is given in MPH and converted to Knots or Mach number:

$$\text{Temperature rise} = (\text{TAS MPH}/100)^2$$


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150 Knots = 2°C

Mach 2.0 = 180°C

250 Knots = 7°C

Mach 3.0 = 390°C

500 Knots = 26°C

Mach 4.0 = 700°C

700 Knots = 55°C

Mach 5.0 = 1180°C

1000 Knots = 100°C

Mach 5.7 = 1530°C

Mach 5.0 by convention is the start of the hypersonic speed range, where the surface temperature of a body (aircraft/re-entry vehicle), has reached approximately 1180°C or higher. Basically, thermodynamics is related to speeds below Mach 5.0) and aerothermodynamics is associated with speeds beyond Mach 5.0.

Aerodynamic heating is also known as the thermal thicket or thermal barrier.

*See Kinetic heating; Impact temperature; Temperature rise.*

**Aerodynamicists** The engineers and scientists who are responsible for the design of aircraft and study of aerodynamics are known as aerodynamicists.

*See separate entries for the following list:*

**Allen, H.; Ackeret, J.; Bernoulli, D.; Betz, A.; Boyle's Law; Breguet, L.; Büseman, A. Dr.; Euler, L.; Glauert, H.; Jacobs, E.; Joukowski, N. Prof; Karman, T.; Küchemann, Dr. D.; Kutta, M.; Lanchester, F. Dr.; Prandtl, Professor L.; Theodorsen T.; Whitcomb, R..**

There are several other aerodynamicists, physicists, and scientists mentioned in this book, who have also contributed to the theory of aerodynamics and the safety of flight.

**Aerodynamic root reference chord** The aerodynamic root reference chord is a term used by aircraft designers and pilots; it is the wing root chord measurement. It is used for example, when referring to the position of the aircraft's centre of gravity when fuel is transferred between fore and aft tanks to maintain the airplane's CG within the prescribed limits.

See **Aerodynamic chord**; **Chord**.

**Aerodynamics** Aerodynamics, a branch of physical science, is the study of the motion of a body (aircraft) moving through the air and the reaction of the air when disturbed by the body, as opposed to stationary air (aerostatics).

The name aerodynamics (aerios dynamis) was taken from the Greek language where *aerios* is associated with the air and *dynamis* means force. The modern term of aerodynamics was first introduced in 1837 in the *Popular Encyclopedia*, which to quote, described 'aerodynamics as a branch of aerology'. However, the subject can be divided into classical, transonic, supersonic, and hypersonic aerodynamics. Propeller and helicopter aerodynamics is both an extension of the above but both have their own additional peculiarities and characteristics. Aerodynamics can be further divided into internal aerodynamics, which studies the airflow for example, through a jet engine and external aerodynamics being the study of airflow around an object (aircraft) which is our main interest here.

However, where did all this information originate? Who made these great discoveries? The science of aerodynamics, some people maybe surprised to learn, started long before aircraft first flew. The beginnings of aerodynamics can be traced as far back to the time of Daniel Bernoulli (1700–1782) and Leonhard Euler (1707–1783) and others. Many other great names have been involved with the development of aerodynamics especially in the first half of the 20<sup>th</sup> Century and can be attributed to a select few – the fathers of aerodynamics.

The early scientists originally referred to classical aerodynamics as fluid dynamics. Classical aerodynamics is based largely on Newton's three Laws of Motion taken from Newtonian mechanics and Bernoulli's Theorem. It deals with subsonic flow with velocities below Mach 0.8, and covers the concepts of air resistance, laminar and turbulent airflow, boundary layers, separation and stagnation points, circulation, Reynolds number and vortices in an incompressible flow, etc. Supersonic aerodynamics assumes the air is a compressible flow and introduces the terms of Mach number, shockwaves, aerodynamic heating and compressibility, etc. Transonic aerodynamics deals with the problems of flight between Mach 0.8 to Mach 1.2 encompassing gradual changes in conditions between the subsonic and supersonic flight regimes.

*See **Aerodynamicists**.*

**Aerodynamic stall** A loss of lift due to the wing's angle of attack being increased up to and beyond the critical angle.

It is also known as a g-stall.

*See **Accelerated stall** and other entries related to **Stall**.*

**Aerodynamic tailoring** The aircraft designer may use aerodynamic tailoring to improve a wing's basic design. The use of leading edge devices, vortex generators, fences, stall strip, trailing edge flaps, wash-out, or varying the section profile along the span can all help to improve the lift characteristics of the wing, reduce drag, and control the stall behavior.

**Aerodynamic theories of propellers** In 1865, the Scottish engineer and scientist William John Macquorn Rankine (1820–1980) founded the propeller's axial momentum theory while working on the theory of ship's propellers. At a later date further work and development on the axial momentum theory, was covered by another engineer, Robert Edmund Froude (1846–1924). The blade element theory was first introduced by William Froude (1810–1979) an engineer and naval architect in 1878, when he also was working on ship's propeller theories. Note the theory of ship and aircraft propellers is virtually the same, because air at subsonic speed behaves very similar to flowing water. Stefan Drzewiecki further developed the blade element theory from 1892 onwards and was credited with the majority of the research work. The axial momentum theory, also known as the Rankine-Froude theory after its two author's deals with the energy change given to the air mass after it passes through the propeller disc. It also includes the effect of the rotational propwash, the friction drag of the propeller blades and the loss of energy in the slipstream caused by the interference of the engine nacelle or the fuselage, amongst other factors. The blade element theory deals with the forces acting on the propeller as it moves through the air at a uniform velocity. It also includes the blade's shape and number of blades and assumes the propeller blade to be made from an infinite number of blade elements, hence the name blade element theory.

**Aerodynamic thickness/chord ratio** The aerodynamic thickness/chord ratio is the ratio of the wing's thickness (depth) to its chord (length).

The thickness/chord ratio of a sweptback wing is the apparent thickness of the wing as the airflow over the wing sees it, which is less than its physical thickness/chord ratio.

See **Thickness/chord ratio**.



**Aerodynamic turning moment** The aerodynamic turning moment is a turning force acting on the propeller's pitch change axis caused by aerodynamic loads. The force tends to turn the propeller towards coarse pitch during cruise flight and towards fine pitch when the propeller is windmilling.

See **Propeller stress**.

**Aerodynamic twist** The aerodynamic twist is the variation of angle of incidence along the wing's span. For most wings, the angle of incidence decreases towards the tip, known as wash-out (the opposite is wash-in but is less common). The reason for aerodynamic twist is to achieve the most suitable lift distribution across the span, usually in the form of an elliptical lift-distribution curve. Twisting the wing ensures the wing root stalls before the tip does. By delaying the stall onset at the tips it helps to keep the ailerons active for roll control. With the wing root stalling before the tip, the stall will be more stable with a positive nose drop, known as a square stall and with less chance of an incipient spin developing.

See **Wing twist**; **Wash-out**.

**Aerodynamic vehicle** Any airplane, helicopter or glider, etc, that is capable of flight in the atmosphere relying on an aerodynamic reaction (lift) to support itself.

**Aeroelasticity** Aeroelasticity involves the study of the aerodynamic forces acting on the aircraft's non-rigid, elastic structure.

The aircraft must be able to flex, or oscillate to a certain degree to ensure safe flight. However, the aircraft's structure must have sufficient stiffness to withstand the various aeroelastic forces imposed during flight and withstand the gust forces due to flight through turbulence. If the structure is too supple, aeroelastic instability can have an adverse effect on the aircraft and torsional flutter can occur in the control surfaces leading to aileron reversal, wing bending, divergence, vibration, and flutter.

When flying through turbulence, the wings may noticeably oscillate up and down as they flex in the turbulence – if they did not do that, it would be cause for concern. It was recognized during the early 1920s when Theodore Theodorsen (1897–1978) of NACA realized that aeroelastic structural flexibility must be considered to ensure safe flight.

*See* **Aileron reversal; Flutter; Flexural aileron flutter; Mass balance; Forward sweep wings; Divergence; Divergence speed.**

**Aeroelastic tailoring** The mission adaptive wing is made of composite flexible materials, which allows on-board computers to change the wing's camber in flight to suit the requirements of the manoeuvre being performed. The wing is designed to flex to imitate the movement of the ailerons, flaps, or leading edge devices. Fighter aircraft using a mission adaptive wing would enjoy greatly enhanced manoeuvrability.

*See* **Mission adaptive wing.**

**Aerofoil** The British favour the term aerofoil as opposed to the American term airfoil. Both terms have the same meaning when referring to wing-shaped objects such as control surfaces, wings or turbine blades, etc.

*See Airfoils.*

**Aerofoil section** Aerofoil section is the term favored by the British.

*See Airfoil section.*

**Aeronautics** The study of the science of flight.

**Aeronautical research facilities** Several aeronautical research facilities have been built, mainly in Europe and North American. A few of the more famous ones where aerodynamic research was and in some cases still is, carried out are listed below:

- **NASA Ames/Dryden Research Centre**, Moffat Field, California, USA
- **NASA Dryden Flight Research Centre**, Edwards AFB, California, USA
- **NASA Langley Research Centre**, Hampton, Virginia, USA
- **NASA Lewis Research Centre**, Cleveland, Ohio, USA
- **Wright-Patterson AFB**, Dayton, Ohio, USA
- **Royal Aircraft Establishment, Farnborough**, England
- **National Physical Laboratory**, England
- **Gottingen Aeronautical Research Institute**, West Germany
- **Volkenrode Research Institute**, Braunschweig, West Germany
- **Institute of Aerodynamics, Aachen**, West Germany.

*See individual entries for above research facilities.*

**Aeroplane/airplane** The term aeroplane, given by Frederick Marriott (c1905–c1984) has the same meaning as the term airplane. The word aeroplane is the preferred British name as opposed to airplane, which is favored by the Americans.

See **Aircraft**.

**Aerophysics** It is the study of aerodynamics at speeds greater than Mach one, (or the speed of sound).

**Aerospace vehicles** The Space Shuttle and the ill-fated Russian Buran types are classed as aerospace vehicles when they are able to maintain controlled flight within and outside the sensible atmosphere. Aerospace vehicles are classed as a sub-group of aircraft; it is incorrect to call them airplanes.

**Aerostatics** Aerostatics is a close relative or subfield of aerodynamics. It deals with the science of gasses at rest (hence static) and in equilibrium, as opposed to aerodynamics – the study of airflow, or planes in motion through the air.

An aerostat is a lighter-than-air craft, which includes free-floating hot-air balloons, airships and dirigibles. The word dirigible is a French word meaning directable or steerable. Aerostats use aerostatic lift that is buoyant in the atmosphere at heights at which it displaces its own mass of air. The word aerostat is taken from the New Latin word of *aerostatica*, which was introduced in 1784.

The pilot of an aerostat is typically known as a Balloonist or an Aeronaut.

**Aerothermodynamics** Aerothermodynamics is the favored term for the combined study of aerodynamics and thermodynamics. At hypersonic flight speeds (greater than Mach 5.0), the real gas effects due to the relationship of high temperatures and mechanical energy must be taken into consideration. It involves the following subsets:

- **Shock layer**
- **Aerodynamic heating**
- **Entropy layer**
- **Real gas effects**
- **Low air density effects**
- **Hypersonic curves**

*See above entries for details.*

**Aft body strake** Strakes mounted on the aircraft's under surface hang more vertically than delta fins. Their purpose is to improve the aircraft's directional stability.

*See **Strakes**; **Chine**; **Vortex lift**.*

**Aft centre of gravity** The aft centre of gravity limit is located at the rearmost position of the centre of gravity range. Longitudinal stability and trim are the major factors that determine the centre of gravity's aft limit. It is a longitudinal stability requirement for the centre of gravity to be ahead of the wing's aerodynamic centre, which is normally located at about 25% MAC.

When the centre of gravity moves towards the aft limit, the reduced tail down force will produce a slightly lower stall speed. However, the airplane can fly more deeply into the stall; deep stalls have an adverse effect on any airplane due to the difficulty of recovery. Because the permitted range of the centre of gravity travel is relatively small on light aircraft, the amount of change to the stalling speed is also relatively small. Stall recovery may be difficult or even impossible if the centre of gravity limit is exceeded

due to the greatly reduced static stability. For a given pitch attitude change, smaller and lighter stick movements are required. With an extreme aft centre of gravity location passed the rear limits, severe control problems may occur detrimental to safe flight. The elevator is required to provide a nose-down force to maintain level flight and eventually the elevator will run out of travel. This will result in a loss of pitch-down control leading to an unrecoverable deep stall.

Recovering from a spin is also more difficult with an aft centre of gravity location; the spin becomes flatter due to the centrifugal force pitching the tail down, and nose up away from the spin's axis. A loaded baggage compartment will only aggravate the spin's characteristics, so therefore baggage is not to be carried when performing spinning exercises. With an aft centre of gravity location, the aircraft is also inclined to pitch, roll, and yaw around more in turbulence making the ride more uncomfortable for the plane's occupants and instrument flying will be more difficult for the pilot to perform.

*See Diagram 12, Centre of Gravity Range. Compare with Forward centre of gravity limit.*

**Aft stagnation point** The aft stagnation point is located at the wing's trailing edge. It initially forms at the start of the take-off run, on the upper surface just forward of the trailing edge. As the aircraft accelerates and the airflow becomes stabilized, the stagnation point/line moves aft to settle at the trailing edge with a stationary (stagnant) region of air. The airflow from above and below the wing rejoins downwind of the stagnation point. This is in accordance with the Kutta-Joukowski condition. A forward stagnation point also exists.

*See Diagram 65, Pressure Differential & Stagnation Points.*

**Aileron drag** The ailerons produce drag when deflected either up or down into the air stream. Drag may be asymmetric being greater on the up-rolling wing where the total reaction is tilted rearwards causing the rearward component to produce increased drag and retard the wing from turning. The opposite happens on the down-rolling wing; the forward component pulls the wing forward in support of the up-rolling wing. The combined forces cause the aircraft to yaw away from the required direction of turn and towards the rising wing. Frise or differential ailerons overcome this problem, although it can still become evident at low air speed with large aileron deflections. Therefore, ailerons produce adverse yaw as opposed to spoilers, which yaw the aircraft in the direction of turn, being pro-yaw.

Aileron drag can assist directional control during take-off in a crosswind. The ailerons should be 'turned into wind' to hold down the up-wind wing during the initial take-off roll; this lowers the aileron on the opposite wing, increasing the aileron drag to oppose the swing into wind. It is more effective on relatively longer span wings.

*See Aileron reversal; Frise ailerons; Differential ailerons.*

**Aileron reversal** An aileron application producing unwanted roll, or control reversal, which rolls the aircraft in the opposite direction to that required; it is particularly evident in high-speed flight increasing with the forward speed squared.

Aileron deflection creates an aerodynamic twisting moment (torsional deflection) on the wing about the OY lateral axis due to the wing's elasticity. An up-deflected aileron will twist the wing's leading edge upwards – the increased angle of incidence (and angle of attack) increases the lift, raising the wing instead of lowering it.

Transport aircraft with swept wings commonly have inboard ailerons to reduce the problem of the wing twisting. By acting on the wing closer to the wing roots – they are less effective, but they cancel the effect of aileron reversal. Spoilers (lift dumpers) also

help to prevent wing twist and aileron reversal. Aileron reversal is a greater problem on high-speed aircraft due to the thinner wing structure having greater elasticity.

Operating under normal flight speeds within the envelope, aileron reversal will not be present. However, above a certain speed known as the aileron reversal speed, divergence may occur causing control difficulties mentioned above. Aileron reversal is a high-speed problem as opposed to adverse aileron yaw, which occurs at low speed. The racing monoplanes of the 1920s were the first aircraft to be afflicted with the problem of aileron reversal followed by the high-speed fighters of WW II.

*See* **Aeroelasticity; Flutter; Flexural aileron flutter; Mass balance; Forward sweep wings; Reversed controls; Divergence; Divergence speed.**

**Aileron reversal speed** Aileron reversal speed is the speed at which aileron reversal begins.

**Ailerons** Ailerons are one of the three primary control surfaces; they provide roll control about the aircraft's OX longitudinal axis. To provide the greatest rolling moment arm, the ailerons are located at the outboard trailing edges of the main wing.

Aileron control force is a function of airflow speed, a change in the wing's upper surface camber due aileron deployment and the Newtonian deflection of the airflow over the wing. A deployed aileron deflects the airflow aft of the wing's centre of lift creating a torque effect about the OY lateral axis; this reduces the wing's angle of attack and its effectiveness. At normal operational flying speeds, this twisting effect is negligible; however, at relatively high speeds, aeroelasticity effects can have a serious consequence, which must be avoided.

Most high-speed jet transport aircraft also have inboard ailerons and/or spoilers, which are less powerful than outboard ailerons to counteract the aeroelastic effect (aileron reversal). The outboard



ailerons maybe locked during high-speed cruise to prevent torsional twisting of the wing leaving the inboard ailerons to provide roll control. The ailerons may also droop a few degrees in unison with the first few degrees of take-off flap on some aircraft types.

Various aviators have made claim to the invention of the aileron, from the Englishman M. P. Bolton in 1868 to the Americans, Glenn Hammond Curtiss (1878–1930) in 1908 who refined the aileron as a control surface and Dr. William Whitney Christmas (1865–1960) in 1914. Robert Esnault Pelterie (1881–1957) introduced the term aileron in 1904, which he used on his glider design and he built the first powered aircraft with ailerons. He also lays claim to the invention of elevons and the joystick control. However, the credit for the modern aileron as we know it today goes to the French-born, Englishman Henri Farman (1874–1958). He used ailerons on all four wings of his Farman *III* biplane, which first flew in April 1909. The Wright brothers saw the advantage of Farman's ailerons over wing warping, which was the usual method of roll control of that era.

Aileron is the French word for little wings.

**Air brakes** Air brakes are used to increase drag with little effect on lift. They are mounted on the wing's upper surface of jet transport aircraft near the trailing edge and just forward of the flaps. When activated they hinge upwards into the airflow to produce extra drag for deceleration in flight and after landing. They may be used in unison for deceleration or in opposition on each wing to assist ailerons in roll control.

The McDonnell Douglas F-18A Hornet and F-15 Eagle both have air brakes installed on the upper fuselage/dorsal position; this location reduces the change in pitching motion when deployed. The British Aerospace BAe 146 has speed brakes, known as diverters installed on the tail cone.

See **Spoilers; Speed brakes; Dive brakes; Diverters; Deceleron; Lift dumpers.**



**Photo:** The McDonnell Douglas F/A-18 Hornet has an airbrake (shown here extended) on the rear fuselage between the two vertical fins.

**Aircraft** The word aircraft generally includes all airborne craft including airplanes, gliders, rotary wing craft, hot air balloons, VTOL craft, and aerospace vehicles if they produce lift and drag during re-entry and controlled decent to landing. The word aircraft is commonly used interchangeably with aeroplane/airplane as this author has done in this work.

In 1850, the American balloonist John Wise (1808–1879) was the first person to use the term aircraft. It came into general use with its present meaning as above, in 1911.

**Aircraft lift coefficient** The aircraft's lift coefficient relates the aircraft's total lift to total wing area, as opposed to the lift coefficient, which relates the airfoil's general lift, planform area and the dynamic pressure.

Note, on some aircraft the fuselage may also contribute to lift generation.

**Aircraft-pilot coupling** *See* Pilot induced oscillation (PIO).

**Aircraft principle axes** An aircrafts frame of reference involving the body or geometric axes.

*See* Axes; Body axes.

**Aircraft wake turbulence** *See* Wingtip vortices.

**Air density** The density of air has a significant effect on aircraft performance and features in the formulas for lift and drag, being part of the dynamic pressure. Air density is determined by air pressure, temperature and the amount of moisture within the air mass. The average air density at sea level is considered to be  $1.225 \text{ kg/m}^3$  or  $0.0765 \text{ lb/cu.ft}$  and varies with altitude and air temperature. Several factors influence the density of the air.

Increased air density is caused by:

- low altitude, increased air pressure, cold and dry air.

Decreased air density is caused by:

- high altitude, decreased air pressure, hot and humid air.

The symbol  $\rho$  (rho) in the lift or drag formula represents it.

See **Density**.

**Airflow** At normal angles of attack, the airflow follows the contours of the wing due to airflow inertia and the pressure gradient. The airflow travels further and faster over the upper surface causing a negative pressure area. Maximum dynamic pressure occurs at the wing's leading edge known as the stagnation point. The airflow over the upper wing surface decreases in pressure with a corresponding relative increase in positive pressure underneath the wing. The difference in pressure between the upper and lower surface is responsible for lift.

At the wing's point of maximum camber and at large angles of attack, the centripetal force of the curving airflow is also at a maximum. With a further increase in angle of attack, the flow will eventually depart the upper surface at a tangent and breakaway to form a turbulent flow with a partial loss of lift (a stall).

At higher angles of attack, the reduced air pressure and increased camber close to the leading edge induces the flow breakaway. Likewise, the airflow moving away from an area of low pressure (see adverse pressure gradient) will readily breakaway into a turbulent

flow and allow higher-pressure air to spill around the trailing edge onto the upper surface. This is known as airflow reversal.

The airflow over an airplane in flight is known correctly as the slipstream.

*See Diagram 65, Pressure Differential & Stagnation Points.*

**Airflow reversal** An airplane flying at high angles of attack into the stall experiences a breakaway of the airflow over the upper portion of the wing, allowing the airflow from under the wing to flow around the trailing edge and forward in the reverse direction over the upper, aft portion of the wing. It is also known as burble.

A helicopter also experiences airflow reversal in its rotor system. As the helicopter's rotor blade retreats towards the rear of the machine, the airflow over the outer portion of the blades will be flowing in the correct direction. However, towards the middle of the blade's span the airflow will be zero and will change to a reverse airflow over the rotor blade's root area.

*See Retreating blade stall.*

**Airfoil loading** The term airfoil loading refers to the pressure distribution over the wing in a chordwise direction (from leading edge to trailing edge).

**Airfoil nomenclature** In the first few years of the 20<sup>th</sup> Century, many airfoils designs were introduced without any form of classification. However, in the late 1930s, NACA introduced an ordered numbering system to identify and classify the families of airfoils available. Their first method used the four-digit numbering system, which indicates the geometric features of the wing, suitable for low-speed aircraft; this was followed later by the five-digit series. The six, seven, or eight digit series are used for high subsonic laminar flow airfoils. The main characteristics all airfoils have in common are their mean line form and thickness, which are related in proportion to the airfoil's chord.

A NACA four-digit airfoil, for example the NACA 4418 (read as forty-four, eighteen) would be deciphered as follows:

- 4 = the maximum camber in hundredths of chord
- 4 = the location of the maximum camber at 40% chord
- 18 = the point of maximum thickness in hundredths of the chord at 18% chord.

**Airfoil requirements** The design of an airfoil must meet certain design criteria to be as efficient as possible for the aircraft's intended use. The following is a list of some of these requirements:

- a low minimum drag coefficient with the least drag at the design lift coefficient
- the maximum lift coefficient is required to be as high as possible for shorter take-off and landing performance
- a high lift/drag ratio
- as small a movement of the centre of pressure as possible
- the maximum pitch moment coefficient is required to be as near to zero as possible
- The airfoil thickness must be deep enough for undercarriage storage and to house the fuel tanks, etc.

**Airfoils** A general name for wings, tail fin, propeller, and helicopter's main and tail rotor blades, turbine engine compressor, and turbine rotor blades.

Airfoils are designed to provide an aerodynamic reaction in the form of lift on the wings, or rotor blades and thrust on propellers. Due to the airfoils shape, it produces an aerodynamic reaction (loosely, lift by suction) over the upper surface.

Airfoils are further classed as symmetrical or asymmetrical (cambered) airfoils. A symmetrical airfoil has both sides of equal curvature, as found for example on the tail fin. An asymmetrical airfoil (cambered) is found for example on most main wings.

**Airfoil section** A theoretical two-dimensional cross-section of the wing cut along the chord line showing the shape (chord and camber) of the wing. Low-speed airfoils have a large radius leading edge and a relatively thick wing at the 25–50% MAC positions before tapering off to the trailing edge. A high-speed or laminar flow airfoil has a sharper, small radius leading edge and thinner airfoil section. Some laminar flow airfoils (known as supercritical airfoils) found on jet aircraft have a relatively flat upper surface with a cusp on the lower surface near the trailing edge, in order to raise the critical Mach number and hence allow faster cruise speeds to be achieved. The airfoil section along the span of the wing may vary from root to tip.

The British favor the term aerofoil section.

See *Diagram 4, Airfoil Terminology*.

**Air load** The air load is the result of the airflow exerting an aerodynamic force on the airplane.

See **Dynamic pressure**.

**Airplane efficiency factor 'e'** The parasite drag and induced drag make up the total drag of the aircraft. The parasite drag coefficient above the zero lift minimum can be included with the induced drag coefficient and is termed the airplane efficiency factor ( $e$ ). This factor will be at some value between 0.7 to 1.0, the value depending on the aircraft's characteristics and whether in a clean or dirty configuration. The aircraft efficiency factor is used in the formula for total drag as shown by the example below:

$$\text{Total drag} = (0.318 C_L^2 / AR e) qS$$

Where 0.318 = constant (reciprocal of 3.14)

$C_L^2$  = lift coefficient squared

AR = aspect ratio

$e$  = airplane efficiency factor

$q$  = dynamic pressure

$S$  = wing area.

Note: the airplane efficiency factor ( $e$ ) is also used in other formulas, notably the Induced Drag Coefficient where the term ' $e$ ' is used instead, for the span efficiency factor.

*See Diagram 18, Drag Bucket; Diagram 20, Total Drag Curves.*

**Air pressure** Air pressure is due to the weight of air pressing down on a unit area surface. The standard sea level air pressure is measured as 1013.25 hPa (mb), 29.92 inHg, 14.696 PSI, or 2116 lb/ft<sup>2</sup>. On the aircraft's altimeter, this setting is also known as QFE.

*See Pressure.*

**Air resistance** This term is more commonly known as drag in aerodynamic terminology.



**Air resonance** Air resonance is similar to ground resonance except, as the name implies, it can occur less commonly, when a helicopter is airborne. A vibratory frequency in the main rotor system can induce the onset of air resonance.

*See Dragging; Ground resonance; Resonance.*

**Airscrew** An alternate name for the propeller is airscrew. The word airscrew was introduced to aviation to distinguish between the aeronautical and marine type propellers (which are referred to as screws). The word airscrew was more commonly used in Europe than in the USA.



**Photo:** The Sopwith Camel sports a wooden airscrew, as did many aircraft of that era. This Camel resides in the National Museum of the USAF, in Dayton, Ohio.

Between the 1920s and 1950s, the name airscrew usually referred to a 'tractor' propeller, which is a propeller mounted in front of the engine as opposed to a 'pusher' propeller behind the engine. It is now an archaic and virtually redundant term and has been replaced by the word propeller or prop for short. However, writers of early aeronautical history quite often use the term airscrew when writing about propellers of that era. The term propeller was first used to describe any mechanical device used to propel a vehicle and it came in to aviation terminology circa 1850 to have the same meaning as the word airscrew, although the term did not fully supersede the term airscrew until circa 1950.

**Air speed** Air speed, as used by pilots and aerodynamicists, refers to the speed of the airflow traversing the aircraft in flight. It may also be written as airspeed (one word).

Air speed is a rather loose term without clarification. The air speed indicator in the cockpit, via the Pitot tube, measures the speed of the air flowing over the aircraft. However, it is subject to errors as the airplane accelerates and changes altitude, due to changing air density, temperature, and compressibility. The air speed indicator is calibrated to ISA sea level conditions with an air pressure of 29.92 inches Hg (1013.25 hPa) and a sea level temperature of +15° Celsius. The indicated air speed over-reads due to ram effect increasing the air density in the Pitot tube. With increasing altitude, the indicated air speed must be converted to equivalent air speed (EAS) to account for compressibility. With increasing altitude, the stalling speed in EAS remains constant while the indicated air speed increases and over-reads. Some aircraft performance charts are drawn up with either indicated, calibrated, equivalent, or true air speed or Mach numbers. The required corrections (normally calculated by pilots on their E-6B navigation computers) are as follows:

- IAS plus instrument and position error = RAS/CAS
- IAS/CAS (RAS) minus compressibility correction = EAS

- EAS plus or minus correction for temperature and density error = TAS.

Where: IAS = indicated air speed

RAS = rectified air speed (UK)

CAS = calibrated air speed (USA)

EAS = equivalent air speed

TAS = true air speed.

The first venturi operated air speed indicator was used by Frenchman Captain A. Etene in January 1911. All air speed indicators are now operated by the Pitot-static system.

*See* **Calibrated air speed; Pitot-static system.**

**Allen, H.** Harvey Julian Allen (1910–1977) was the Director of NASA Ames Research Centre from 1965 to 1969. He is best remembered for his introduction of the blunt nose aerodynamic theory for aerospace vehicles. A blunt nose diffuses much of the heat generated during re-entry to the Earth's atmosphere.

Fellow scientist, Alfred J. Eggers Jr. (1922–2006) assisted him.

*See* **Blunt nose re-entry vehicle.**

**All-flying tailplane** An all-flying tailplane is one with a variable-incidence – not fixed as on a conventional tailplane/elevator. All-flying tailplanes are more powerful than a conventional tailplane/elevator system and can produce greater pitching forces. It has an anti-balance tab to resist movement of the tailplane. A pull on the control column by the pilot will raise the trailing edge of the tailplane to pitch the nose upwards. The anti-balance tab will also rise up to increase the pounds of pull required, thus preventing an over-control situation from developing.



**Photo:** The Piper PA28 Cherokee line of aircraft all use an all-flying tailplane. This photo shows the tailplane in the raised (trailing edge up-position) with the balance tab, which also rises (and descends) with the tailplane movement to increase the pounds of pull required from the pilot.

Robert Rowe Gilruth (1913–2000) of NACA invented the all-flying tailplane to reduce the effects of Mach tuck on early jet fighter aircraft. The Curtiss XP-42 (a modified P-36 single, piston-engine fighter) was used to flight test the first all-flying tailplane.

It is also known as an all-moving tailplane or a stabilator.

See **Slab tailplane**; **Stabilator**.

**All-moving tailplane** *See All-flying tailplane.*

**Alpha  $\alpha$**  An alternate term for angle of attack; used when referring to the main wings in particular but it can also be applied to the tailplane and tailfin. The Greek letter alpha ( $\alpha$ ) denotes the angle of attack.

**Altitude pressure ratio** The altitude pressure ratio is the ratio of the ambient static pressure ( $p$ ) to the standard sea level static pressure ( $p_o$ ). The altitude pressure ratio is allotted the Greek symbol of  $\delta$  (delta). This ratio is commonly used in jet engine performance.

$$\text{Altitude pressure ratio} = \frac{\text{Ambient static pressure}}{\text{Standard sea level static pressure}} = p/p_o$$

*See Static air pressure.*

**Ames, Dr. J.** Dr. Joseph Sweetman Ames (?–1943) was a founding member of NACA in 1915. Amongst many other endeavors, he directed NACA into aerodynamic research. In 1927, he became the chairperson of NACA's main committee. With WW II on the horizon, he was instrumental in gaining funding for a second research laboratory and more NACA facilities.

In recognition for Dr. Ames's influence in NACA, the new laboratory at Sunnyvale, California was named after him on 8 June 1944 as the Ames Aeronautical Laboratory (now known as the Ames Research Centre).

**Ames/Dryden Research Centre** The Ames/Dryden Research Centre is located at Moffett Field, Sunnyvale, forty miles south of San Francisco, California. The facility takes its name in honor of Dr Joseph S. Ames, a leading aerodynamicist working for NACA. The centre was originally named the Ames Aeronautical Laboratory when first established in 1944 for NACA. In 1958, its name was changed to the Ames/Dryden Research Centre, operated since then by NASA.

Ames/Dryden Research Centre was the second laboratory (after Langley) to be built for NACA, which opened on 8 June 1944. Several wind tunnels have been installed there for various research projects over the years, including research into aircraft structures, development work, obtaining new knowledge and technology including the National Full-Scale Aerodynamic tunnel for testing full size aircraft. It is now NASA's primary research facility along with its flying base at Edwards in California. The other two bases are Langley Research Centre in Virginia and Lewis Research Centre, renamed the John H. Glen Research Centre at Lewis Field in Cleveland, Ohio.

The NASA Space Shuttles were also tested at the Ames Research Centre along with other manned space vehicles.

See **Wind tunnels**.

**Angle of advance** The angle of advance of a propeller blade is related to the helix angle. *Diagram 68, Propeller Pitch*, shows the helix angle AB-AC.

**Angle of attack ( $\alpha$ )** The angle between the free stream relative airflow (RAF) and the wing's chord line is the angle of attack. Stated simply, it is the angular difference between where the wing is *going* and where it is *pointing*. The Greek letter alpha ( $\alpha$ ) is the symbol used for angle of attack, which refers to the whole wing, as opposed to the section angle of attack of an airfoil section, denoted by the symbol of  $\alpha_o$ . The angle of attack ( $\alpha$ ) is correctly termed the geometric angle of attack; unless otherwise defined, reference is always to angle of attack ( $\alpha$ ).

The absolute angle of attack is the angle between the chord line and the angle of zero lift.

See **Diagram 4, Airfoil Terminology; Induced angle of attack ( $\alpha_i$ )**.

**Angle of bank ( $\phi$ )** The angle of bank refers to how steeply the wings are tilted from the horizontal during a turn.

The angle of bank is independent of aircraft weight and there is only one bank angle for a given radius of turn and air speed. An increase in the speed requires an increase in bank angle to maintain the same radius of turn and vice versa. The angle of bank symbol is given the Greek letter of phi ( $\phi$ ). The required angle of bank is found from the following formula:

$$\text{Angle of bank} = WV^2/gr \div \text{weight} = \tan \theta = V^2/gr$$

Where  $V^2$  = aircraft speed in meters/second<sup>2</sup> (FPS<sup>2</sup>)

W = weight

g = gravity at 9.81 m/s<sup>2</sup> (or 32.2 FPS<sup>2</sup>)

r = radius in meters (feet).

See **Rate of turn; Radius of turn; Diameter of turn; Turning.**

**Angle of climb ( $\gamma$ )** The aircraft's angle of climb is important when obstacles are to be cleared after take-off. The maximum angle of climb air speed ( $V_x$ ) is slower than the maximum rate of climb speed, and allows the aircraft more time to climb and gain the maximum height for the shortest distance flown. The maximum angle of climb is dependent on excess thrust in *pounds*, the greatest difference between thrust required and thrust available, as opposed to the rate of climb, which is related to excess thrust *horsepower*. The formula for the maximum angle of climb is:

$$\text{Maximum angle } (V_x) = \sin \gamma = \frac{T-D}{W} = \frac{D}{W}$$

Where T = thrust

D = drag

W = weight.

*See Climb performance; Diagram 60, Thrust & THP Required & Available Curves.*

**Angle of incidence** The angle of incidence is the angle between the airfoil's chord and a reference line on the airplane's OX longitudinal axis. It may be referred to as the rigger's angle of incidence or the geometric angle of incidence. The wing's angle of incidence usually changes from the wing root to the wingtip, resulting in wing twist, or wash-out.

Note the root angle of incidence is fixed relative to the airplane's longitudinal axis, but the angle of attack varies in flight. The two terms should not be interchanged.

It is also known as the wing's angle of incidence, or wing incidence.

**Angle of inclination** *See Angle of incidence.*



**Angular velocity & acceleration** A body (aircraft) will always obey Newton's 1<sup>st</sup> Law of Motion; 'it will continue moving in a constant direction at the same speed unless acted upon by another force'. Therefore, to make an airplane change direction requires an additional inertial force to provide the acceleration towards the centre of the turn.

We normally associate acceleration with an increase of speed. However, when an aircraft turns at a constant speed, acceleration is still present due to a change in direction. Acceleration is a vector quantity having both direction and magnitude. Therefore, although the speed remains constant, the direction changes during the turn. The change in velocity requires an acceleration of magnitude equal to:

$$\text{Acceleration} = V^2/R$$

Where  $V^2 = \text{m/s}^2$  (or ft/sec<sup>2</sup>)

R = radius of turn in meters (or feet).

Angular acceleration is also known as centripetal acceleration.

*See* **Turning**.

**Anhedral** Anhedral is the negative dihedral angle between the transverse horizontal reference plane and the downward inclination of the wings from the wing root to the wing tip. The wings appear to sag downwards under the effect of their own weight. Some high wing airplanes have anhedral; the purpose is to reduce the inherent pendulum stability of the high wings to improve controllability.



**Photo:** The Boeing C-17 Globemaster III cargo transport has anhedral wings for stability purposes.

The BAe 146 jet transport aircraft is one example with main wing anhedral, while the McDonnell F-4 Phantom multi-role combat aircraft has tailplane anhedral. The BAC TSR.2 (Tactical Strike and Reconnaissance) jet bomber prototype had fixed, anhedral wingtips on its main wing drooped at about forty-five degrees. Anhedral is also known as cathedral (pronounced as cat-hedral).

*Compare with **Dihedral angle**.*

**Annular wing** *See* Flat Annular wing.

**Anti-balance tab** The anti-balance tab provides an increased stick force by moving in the same direction to the elevator's movement.

The anti-balance tab works to provide greater control feel and prevent over-controlling and over-stressing the elevator by requiring more pounds of pull from the pilot. The anti-balance tab is attached to the elevator's trailing edge and will move in unison with it; if the elevator's trailing edge is raised, the anti-balance tab will also move up to provide an opposing downward pressure on the elevator and vice versa.

It is known as an anti-servo tab when mounted on a stabilator.

*See* Trimming devices.

**Anti-servo tab** *See* Anti-balance tab.

**Anti-shock bodies** Anti-shock bodies are located on the trailing edge of the main wings of some large transport aircraft. They smooth out the airflow over the wing and act as an area rule type of device to control the boundary layer over the wing's rear portion.

The trailing edge flaps have their flap tracks protruding into the air stream causing drag, therefore streamlining the flap tracks with flap track fairings has the added advantage when they double as anti-shock bodies.

*See* Flap track fairings; Küchemann carrots.

**Anti-spin strip** *See* Stall strip.

**Anti-torque tail rotor** A tail rotor is mounted on the end of a helicopter's tail boom to oppose the main rotor torque and to provide directional control in the yawing plane.

A helicopter with the main rotor turning anti-clockwise has its tail rotor drawing air from the right side of the tail boom and accelerating the flow as it passes through the rotor to exit on the left-hand side – in the same way a propeller works. Varying the amount of anti-torque rotor thrust controls the helicopter in the yawing plane.

*See* **Tail rotor**.

**Apparent thrust** Apparent-thrust is the term used to describe the airflow interference due to the location of the engine nacelle or fuselage on the propeller's free thrust. Free thrust assumes the propeller to be free standing without any disturbance to the airflow passing through it.

Also known as gross thrust.

*See* **Freestanding propeller; Thrust terminology**.

**Aperiodic stability** The aircraft is said to have aperiodic stability, which is strongly damped and causes the aircraft to return to its trimmed speed without any oscillations. Aperiodic means to have no period.

*See* **Stability**.

**Area increasing flaps** Large airplanes are equipped with trailing edge flaps that move rearwards as they deploy downwards to increase the wing's area. Thus, more lift is produced allowing lower take-off and landing speeds.

They are also known as Fowler flaps or extension flaps.

**Area ratio** The area ratio of the wing is defined as the wing area divided by the wingspan squared ( $S/b^2$ ). Area ratio is the reciprocal of aspect ratio.

*Compare with* **Aspect ratio**.

**Area rule** Richard T. Whitcomb (1921–2009) in 1953 discovered the fact that indenting the fuselage at the wing root would reduce shockwave drag at the fuselage/wing junction. He was credited with this discovery, and received the Collier Trophy in 1954 during his work at NASA Langley. Although Richard Whitcomb received the credit for introducing the area rule concept, Dr. Dietrich Küchemann (1911–1976), the German aerodynamicist, had previously addressed the idea. During the 1930–1940s in Germany, he pioneered the area rule concept when studying supersonic airflow problems. However, compared to Richard Whitcomb, Dr. Küchemann followed a different line of study when he investigated reducing the effect of the spanwise flow along the wing, rather than reducing the drag itself. It is also claimed Otto Frenzl (1909–1996) while working on the Junkers transonic wind tunnel, discovered the area rule concept and received a German patent in 1944 for his research, which he called the bottle rule.

Early first generation transonic and supersonic jet fighters, suffered from the high interference drag rise in the area of the wing root and fuselage intersection. This was due to the rapidly increasing thickness of the wing root causing shockwave drag in this area. To reduce the drag rise when flying in the transonic region approaching Mach 1.0, area rule was introduced to solve the problem.

The total cross-section area of the fuselage is increased and decreased along the aircraft's OX longitudinal axis forming a waist or Coke bottle shape in the wing root area. The fuselage cross-section follows a smooth and gradual decrease followed by a gradual increase aft of the 'waist'. This modification reduces the strength of the shockwaves in the transonic region. Area ruling the fuselage, especially where the wing's camber is greatest in depth, reduces the shockwave formation and its associated drag.

Corporate jets now commonly have the area between the aft fuselage-mounted engines and the fuselage itself indented (area ruled) for this purpose. In addition, very long fillets serve the same purpose underneath the fuselage on these jets.

The delta wing Convair F-102 Delta Dagger jet fighter was first built without area rule and did not perform as well as expected and was unable to reach its intended supersonic speeds. This is where Richard Whitcomb enters the story to solve the problem; he realized shockwave drag was the cause and the area rule concept was born. The Convair F-102 being the first aircraft to incorporate area rule in its final design, and it went on to perform successfully as originally intended.

It is interesting to note, Concorde does not have, or require area rule – the fuselage is not long enough in comparison to the wing root chord! Therefore, the fuselage sides are straight and devoid of any convex curves or area ruling.

It is also known as transonic area rule and Whitcombe area rule. Other names include coke bottle or wasp-waisting.

See **Hayes, W. D.**



**Photo:** A delta wing drone showing its area rule planform. This photo was taken at the Museum of the US Air Force in Dayton, Ohio.

**Arm** The term 'arm' is used in weight and balance calculations. It is the horizontal longitudinal distance between a reference datum on the aircraft and its centre of gravity. The arm is a positive value when measured aft of the datum and a negative value for forward measurements.

It is also known as the moment arm.

**Arrested propeller system** An arrested propeller system is a brake used to stop the propeller from rotating on turboprop-engine airplanes when the engine is left running after flight.

This is possible on a free-shaft turbine engine because the compressor/turbine (gas generator) is mounted on its own shaft. It supplies a flow of hot exhaust gas pressure to the power turbine attached to the propeller shaft via the reduction gearbox. There is no direct mechanical link between the power turbine and the gas generator, hence the name of free-shaft turbine engine. Therefore, during times when a quick turn-around is required between trips the propeller can be stopped by use of the prop brake. This is a safety precaution for personnel who may be moving around near the aircraft while the engine remains running under power.

**Arrow/swept wing** Theodore von Karman (1881–1963) used the term arrow wing in reference to what is now known as a swept wing, which became the accepted term since 1948.

*See* **Sweptback wings.**

**Articulated rotor** A helicopter's articulated rotor hub has three axes of movements where each individual blade is allowed to drag, flap and change pitch angle. The use of drag hinges, flapping hinges and the pitch-change mechanism allows full and free movement of the blades.

Juan de la Cierva's (1895–1936) United Kingdom firm – the Cierva Autogiro Company, first developed the three axes articulated hub. With this new invention, it led the way for the development of the helicopter, as we know it today. Cierva's early Autogiros no longer had the need for ailerons on short stub wings to control roll.

*Compare with* **Semi-rigid rotor; Rigid rotor.**

**Aspect ratio** The aspect ratio (AR) is a measure of the slenderness or fineness ratio of the wing, or the wing's proportions. It has its importance in aerodynamic characteristics of the wing and in structural weight analysis for the aircraft designer. The wing planform has more influence than does the wing area. The tip-to-tip wingspan (b) along with its chord (c) determines the aspect ratio, which is defined as the ratio of wingspan to average wing chord (span/chord) or more conveniently as (span<sup>2</sup>/wing area). This can be shown by either formula:

$$AR = b/c \text{ or } AR = b^2/S$$

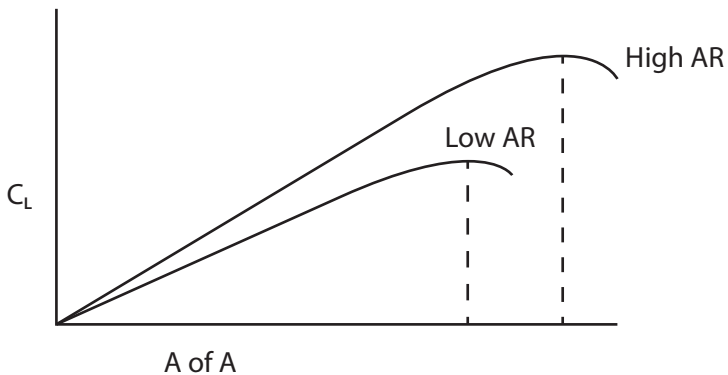
Where AR = aspect ratio

b = span

c = chord

S = wing area.

When making calculations for rectangular wings, the first ratio (b/c) is used; for other planform wings, the second ratio (b<sup>2</sup>/S) is more convenient.



**Diagram 5, Aspect Ratio & Lift Coefficient**



A high aspect ratio wing has the following characteristics:

- a narrow chord, which is more efficient on subsonic aircraft
- the downwash is less and the strength of the trailing edge and wingtip vortices are reduced
- the induced drag is reduced which increases the wing's lift/drag ratio and efficiency
- it creates more lift near the fuselage and is therefore less affected by wingtip vortices
- High moments of inertia, which reduces the roll rates – an advantage for jet transports, but not so for jet fighters.

The opposite effects occur for low aspect ratio wings.

The aspect ratio for a propeller or rotor blade is determined by the ratio of propeller radius to propeller mean chord. A high aspect ratio propeller blade has the same characteristics and advantages as a high aspect ratio aircraft wing. **Diagram 6, Aspect Ratio & Vortex Drag** shows this.

Reference to **Diagram 5, Aspect Ratio & Lift Coefficient**, it can be seen that a high aspect ratio wing produces a greater lift coefficient and is therefore more efficient than a low aspect ratio wing for normal subsonic flight. General aviation aircraft types are more suited to wings with a medium aspect ratio. An aspect ratio of seven to eight is common on subsonic jet transport aircraft and a ratio of two to three is found on jet fighter aircraft. For supersonic airplanes a low aspect ratio, delta type wing is more suitable to improve high-speed efficiency.

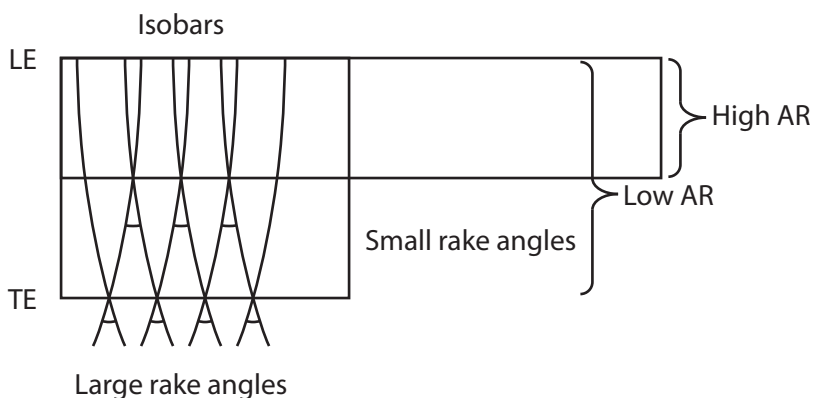
The free stream airflow above and below the wing meets at the trailing edge at an angle as a sheet vortex to flow off each wing, which modifies the original free stream flow direction to angle downwards – this is known as the induced downwash. Long span wings have their tips further apart and the downwash behind the wing has less effect on the airflow. In addition, low aspect ratio wings have, of course, less span distance between the wingtip vortices, which bring them closer together and intensifies the downwash vortex sheet. This causes a reduction in the effective

angle of attack. The aircraft will then be required to fly more nose-up (higher angle of attack) than a high aspect ratio wing for the same lift. Greater lift per unit span (as found on a low aspect ratio wing) produces greater downwash and increased induced drag.

The rake is the angle between the isobars (the lines representing the airflow across the wing) and is less at the wing roots and increases towards the wingtips; it would be a zero rake angle at the centreline if the fuselage were absent. The rake angle is greater for a low aspect ratio wing. An aspect ratio of around eight produces a trailing edge rake angle of approximately  $20^\circ$ . A lower aspect ratio of five will produce a rake angle of around  $25^\circ$ . Therefore, higher aspect ratio wings have smaller rakes and hence, weaker vortices and less induced drag and greater efficiency in comparison to low aspect ratio wings.

The Englishman Frances Herbert Wenham (1824–1908) introduced the term of aspect ratio as early as 1866. Professor Ludwig Prandtl's experimental data proved when he developed his lifting-line theory, that the induced drag is inversely proportional to the aspect ratio.

See **Ground effect; Span efficiency factor; Infinite wing; Rake angle; Diagram 38, Ground Effect Changes.**



**Diagram 6, Aspect Ratio & Vortex Drag**

**Asymmetric airfoil** An asymmetric airfoil is one that has a cambered upper surface as found on propeller blades and most, but not all airfoils. Some helicopter rotor blades are also asymmetric airfoils.

It is also known as a cambered airfoil.

See *Diagram 11, Camber & Lift Coefficient; Cambered airfoil.*

**Asymmetric blade effect** See **Asymmetric disc loading.**

**Asymmetric disc loading** The down-going propeller blade operating at a greater angle of attack due to the inclined propeller axis and the variation in propeller tip speed around the disc, (although the RPM remains constant) causes asymmetric disc loading and uneven thrust over the propeller disc.

Also called asymmetric blade effect.

See **P-factor.**

**Asymmetric flight** A multi-engine aircraft flown with one or more engines shutdown due to failure with continued flight being maintained on the remaining engine(s).

**Asymmetric stall** An asymmetric stall is an undesirable stall condition where one wing stalls before the other. One wing is at a relatively small angle of attack and producing lift and the other wing is at a greater angle of attack and stalled producing greater drag and yaw. This is the recipe for a spin to develop. Hence, an asymmetric stall is to be avoided at all costs.

See **Square stall; Spinning; Stall.**

**Asymmetric thrust** On a conventional twin-engine airplane with both propellers turning clock-wise, (rotating to the right when viewed from behind the propeller), asymmetric thrust causes the greatest yaw when the left-hand engine is shutdown. This is due to greater thrust being generated on the down-going side of the propeller disc.

On a right-handed propeller (rotating to the right), this places the centre of thrust to the right of the propeller axis. On a conventional twin-engine aircraft (without counter-rotating propellers), the centre of thrust on the left-hand engine will be closer to the aircraft's OZ normal axis. However, on the right-hand engine, the centre of thrust will be further away from the axis. In other words, the centre of thrust is offset to the right of the propeller's axis on both propellers.

In the event of a left-hand engine failure, the right-hand engine will cause the greatest yawing moment. Not forgetting, the drag of a windmilling propeller will also contribute to the yawing force. In the case we are considering here, the left-hand engine is said to be the critical engine, due to the greater yaw force and increased handling problems involved. Aircraft with propellers rotating anti-clockwise, or left-handed propellers, will have their right-hand engine as their critical engine. It should be obvious here, that control problems are more critical when the critical engine is shut down.

**Atmosphere** The atmosphere surrounds the Earth as a gaseous envelope (commonly known as air) the nearest layer of which is the troposphere separated from the next layer, the stratosphere, by the tropopause. The stratosphere is the domain for jet aircraft. It is within the troposphere where most aircraft operate and where the majority of weather systems are found.

The top of the atmosphere is ill defined, but is considered by convention to be at an altitude of 100 000 feet, where the sky turns from a blue colour to black. An alternate measurement accepted by the FAI, is the Karman line, named after Theodore von Karman (1881–1963) where the boundary between aeronautics and astronautics (or the boundary of the Earth's atmosphere and outer space) is considered to be at 100 km (62 statute miles).

The term atmosphere is from the 17<sup>th</sup> century New Latin word *atmosphæra* derived from the Greek 'atmos' (meaning 'vapor') and *sphaira* (meaning sphere).

See **International Standard Atmosphere; Structure & composition of the atmosphere.**

**Attached shockwave** An attached shockwave as the name suggests, is attached to some 'sharp' part of the aircraft. For most aircraft, an attached shockwave is preferable to a detached bow shockwave that has moved ahead of the aircraft. The bow shockwave may start as an attached shockwave and then becomes detached from the aircraft's nose or wing leading edge with increasing speed. It depends on the design of the vehicle – blunt or sharp nose – and its speed. In order to keep the shockwave attached a sharp nose angle is required (as found on Concorde and other supersonic aircraft). These types of aircraft usually have wings with sharp leading edges for the same purposes.

See **Detached shockwave; Blunt nose re-entry vehicle.**

Compare **Diagram 8, Detached Normal Shockwave.**

**Attinello flaps** The Lockheed F-104 Starfighter was the first operational aircraft to employ Attinello flaps. They are blown flaps using high-pressure engine bleed air to direct a stream of air over the flaps from a slot on the wing's upper surface ahead of the flaps, which allows the flaps to be deployed at a greater angle without flow separation. John D. Attinello from the US Naval Air Test Centre invented the flaps named after him.

*See Blown flap.*



**Photo:** The Lockheed F-104 Starfighter was the first operational aircraft to employ Attinello flaps. The Starfighter shown here was viewed at the Midland Air Museum at Coventry, England.

**Attitude** The aircraft's attitude in flight is described with reference to the three major axes of OX longitudinal axis, the OY lateral axis, and the OZ vertical axis.

*See Axes.*

**Augmented oscillations** *See Pilot induced oscillation (PIO).*

**Autogyro** An autogyro is a very efficient STOL aircraft powered by an engine/propeller combination for forward thrust and a rotor blade system for lift, which rotates under the influence of the craft's forward motion through the air. The rotor system is a semi-rigid, two-blade system; however, unlike the helicopter, the autogyro's main rotor system is un-powered, and has zero engine torque, therefore a tail rotor is not required.

The relative airflow (RAF) through the rotor disc is due to the blades rotation and the craft's forward speed. The relative airflow enters the rotor disc from below, producing constant autorotation of the main rotor, as opposed to entering from above the disc as it does on a helicopter in normal flight. The resultant force is perpendicular to the relative airflow and is divided into the forces of lift and blade drag. Because the resultant is tilted forward of the axis of rotation, the forward component of the resultant force



**Photo:** This McCulloch Super J-2 autogyro is displayed in the Pima Air & Space Museum, Tucson, Ar, USA.

drives the rotor blades continuously in autorotation. The forward component is equal and opposite to the rotor blade drag.

Juan de la Cierva (1895–1936) from Spain initially developed the autogyro (spelled as Autogiro) as an alternate idea to the helicopter. His first successful machine was the Cierva C4 Autogiro, which on 9 January 1923 made the first officially recorded successful flight of an autogyro in Madrid, Spain. [Note the difference in spelling between Cierva's Autogiro (its trade name) and the autogyro in general]. The name autogyro is taken from the verb 'gyrate', which means to rotate as a windmill does.

*See* **Cierva, Juan de la.**

**Automatic slat** An automatic slat is held in the open position away from the wing's leading edge at low speeds; increasing air speed (and dynamic pressure) forces the slat closed against the wing's leading edge. With increasing angle of attack and reducing air speed, the stagnation point moves to a lower point on the leading edge. The decreasing air pressure on the upper surface allows the force of the spring to push the slat open again. When the slat is open, the air flowing through the slot between wing and slat intensifies the lift force allowing the aircraft to fly safely at lower speeds.

*See* **Slats.**

**Autorotation – airplane** The stabilized autorotation of an airplane is aerodynamically the same manoeuvre as a fully developed spin. During the autorotation (spin), the airplane is continuously yawing, rolling and pitching. The nose attitude may be either well down – the preferred attitude in a spin – or relatively horizontal, which is undesirable due to the difficulty of recovery.

Spinning is the preferred name here to differentiate from a helicopter's autorotation, described below.



**Autorotation – helicopter** A helicopter can perform a controlled descent to a safe landing in autorotation mode, following an engine failure.

During an autorotational descent, the induced inflow air entering the rotor disc from below drives the rotor blades around and produces lift in the same manner as wind drives the blades of a windmill around. Albeit the helicopter is descending at quite a high rate of decent, it is still under control and a safe landing should ensue.

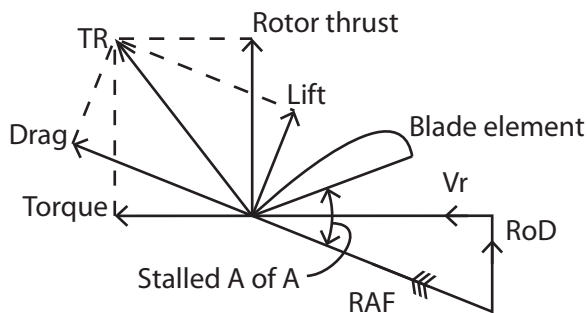
The descent inflow has a different aerodynamic reaction on three different areas of the rotor blade. These three areas are known as the stalled region located at the rotor blade root (*Diagram 7A*), the driven or propeller region at the blade tip (*Diagram 7B*), and the driving or autorotational region covering most of the blade between the tip and root areas (*Diagram 7C*).

The stall region (*Diagram 7A*) is located at the rotor blade root area. [In normal powered flight, stalling may occur on the retreating blade tip area, but it is the root area that is stalled during autorotation]. Due to the relatively high rate of descent, the relative airflow enters the rotor disc from below, which results in a large angle of attack at the blade root. The lift vector is tilted well forwards of the axis of rotation (it is always at right angles to the relative airflow). The rotor drag is high due to the high angle of attack causing the total reaction to lean well back, which causes increased rotor torque, and retarded rotor RPM. The lift/drag ratio is not very efficient in the blade root region.

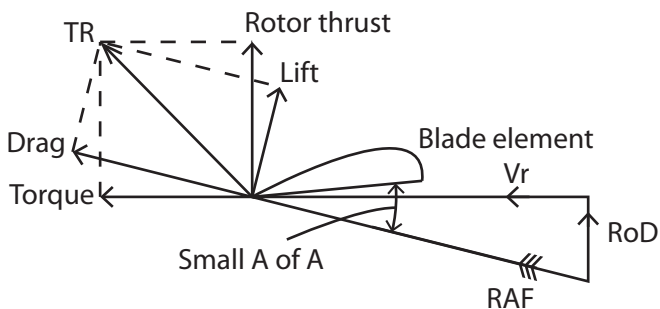
The driven or propeller region (*Diagram 7B*) is located at the rotor blade tip area, where the rotational velocity is very high, and combined with the inflow velocity, the angle of attack is small. The total reaction is angled well to the rear away from the axis of rotation, which produces a very poor lift/drag ratio. The rotor thrust reduces and rotor torque increases considerably, which retards the rotor RPM.

The driving or autorotation region (*Diagram 7C*) is the largest of the three areas and covers most of the rotor blade between the

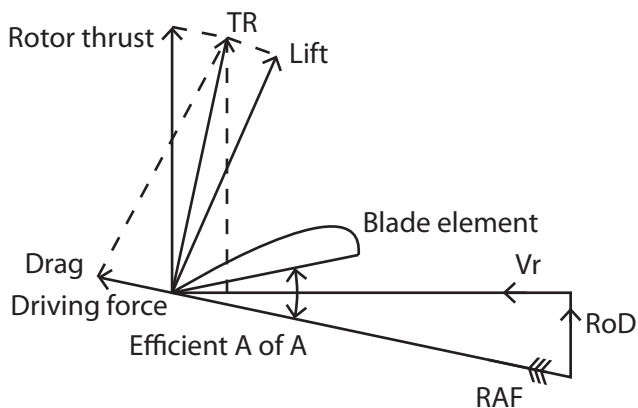
7A



7B



7C



**Diagram 7, Autorotation Forces – Helicopter**

blade tip and blade root areas. The driving area creates forward rotor thrust, which propels the blade around to provide lift to control the helicopter's rate of descent. As found in the stalled root and blade tip areas the inflow comes from below the rotor disc at such an angle the lift vector is tilted forwards and provides a good amount of autorotative lift due to the relatively efficient angle of attack. The torque vector acts in a forwards direction, which drives the rotor blade to produce autorotative thrust.

The rotor blade tips and roots have the rotor torque forces acting rearwards, the same as in normal flight. Therefore, the retarding torque force of the rotor blade tips and blade roots is balanced by the equal and opposite autorotative thrust force – after an initial increase in RPM, the blades continue to autorotate at constant RPM.

The requirement is for as large a lift/drag ratio over as much of the rotor blade as is possible for efficient operation. The steady state rotor RPM depends on the location of the total reaction with respect to the axis of rotation, which is determined by the lift/drag ratio acting on the rotor blade tip and root areas and in between them.

In a theoretical vertical autorotation descent, the three regions are concentric circles around the rotor mast. However, an increase in forward speed changes this pattern off centre towards the retreating blade side. The retreating blade's stalled region increases while the driven region at the tips is reduced. The driving region remains unaltered, albeit it is moved outboard. On the advancing blade, the opposite occurs; the blade root area decreases with the tip area increasing. Again, the driven region remains unchanged.

The angle of attack and hence, lift coefficient is increased on the retreating blade and decreased on the advancing blade, due to blade flapping – dissymmetry of lift and blowback are forever present. These changes in angle of attack accounts for the increased and decreased stall regions on the retreating and advancing blades respectively.

*See **Total rotor thrust; Autorotative thrust or force.***

**Autorotative lift** Autorotative lift is the lift force produced by the rotor blades when the helicopter is descending in autorotation following a power loss. The autorotative lift is in line with the axis of rotation and the total rotor thrust is inclined forwards.

See **Total rotor thrust**; *Diagram 7, Autorotation Forces – Helicopter*.

**Autorotative thrust or force** In autorotation, the aerodynamic total reaction is tilted forward of the axis of rotation and the horizontal component is now acting in a forward direction. It pulls the blades around instead of retarding them (as drag in normal powered flight). Therefore, the rotor blades continue to rotate without engine power during an autorotation descent.

See *Diagram 7, Autorotation Forces – Helicopter*.

**Average chord** The average chord is an alternate name for the geometric chord. It is used to find the wing area by multiplying the average chord by the wingspan.

**Average relative wind** Air flowing over the wing has an induced downward component called the average relative wind. The angle between the wing's chord line and the average relative wind component is the section angle of attack.

See **Section angle of attack ( $\alpha_o$ )**; *Diagram 43, Induced Drag*.

**Avogadro's law** Amadeo Avogadro (1776–1856) an Italian physicist introduced his law named for him in 1811, but it was to be another fifty years before his law was officially recognized. The law states an 'equal volume of ideal gas at constant temperature and pressure contains the same number of particles or molecules'. Avogadro also coined the term molecule meaning a collection of atoms. The law is written as:

$$V/n = k$$

Where:  $V$  = volume of gas

$n$  = number of molecules in gas

$k$  = constant.

Avogadro's law has its significance in aerodynamics with reference to wing lift and engine power. Air with water vapor present (as in flying through rain or cloud) is less dense than dry air and therefore, produces less wing lift force. Likewise, moist air flowing through the engine will produce less power than the same volume of dry air. This is due to the water vapor molecules present in the air mass replacing some of the oxygen and nitrogen molecules, which are reduced by a like amount to maintain constant temperature and pressure.

Moist air has a molecular mass of 18 g/mol, which is less than the dry air molecular mass of approximately 29 g/mol; hence, moist air is less dense than dry air. Therefore, the number of molecules in a specific volume of gas at the same temperature and pressure will observe the ideal gas law. In normal ambient air, (which consists of various gasses) the law is approximate, but relevant for the purpose stated above.

Avogadro's law further defines the ideal gas law by the inclusion of the molecular weight of the gas.

**See General Gas Law; Charles' and Gay-Lussac's Law; Gay-Lussac's Law; Ideal gas law.**

**Axes** An aircraft in flight moves in three dimensions around its centre of gravity (O origin). Its flight attitude is referenced to the three axes, the OY lateral, OX longitudinal and OZ vertical axis, about which the aircraft pitches, rolls and yaws respectively. In accordance with the rules of geometry, the OX longitudinal axis is positive when the aircraft rolls to the right; the OY lateral axis is positive when the aircraft pitches nose up; and the OZ vertical or normal axis is positive when the aircraft yaws to the right. The following table lists the three axes, direction of positive movement, and the primary and secondary effects of aircraft movement. Note, the word axes is the plural and axis is the singular.

The axes system was devised in 1903 by G.H. Bryan.

| Axis            | Direction  | Primary effect | Secondary effect |
|-----------------|------------|----------------|------------------|
| OX Longitudinal | Roll right | Roll           | Yaw              |
| OY Lateral      | Pitch up   | Pitch          | Air speed        |
| OZ Vertical     | Yaw right  | Yaw            | Roll             |

See **Aerodynamic axes; Lateral axis; Longitudinal axis; Normal axis.**

**Axial momentum theory** The axial momentum theory, also known as the Rankine-Froude theory after its two author's deals with the change of thrust or lift energy given to the air mass after it passes through the propeller disc or the helicopter's rotor disc. It also includes the effect of the rotational slipstream, the friction drag of the propeller blades and the loss of energy in the slipstream caused by the interference of the engine nacelle or the fuselage, amongst other factors.

The axial momentum theory deals with the air mass well ahead of the propeller disc, which is assumed stationary. On approaching the propeller disc and passing through it, the air mass accelerates and gains momentum. Thrust is a result of this changes in momentum and the greater the change, the greater will be the

thrust. Momentum is the product of mass multiplied by velocity (MV). A large mass of air flowing at a small velocity can produce the same amount of thrust as a small mass of air flowing at a high velocity. The former, a large mass at a low velocity is the better of the two because it requires less work to produce the same amount of propeller thrust. The following can show this; the slipstream gains kinetic energy ( $\frac{1}{2} MV^2$ ) due to the increase in momentum (MV) as it passes through the propeller disc. The kinetic energy represents the work done by the propeller in accelerating the air mass from zero velocity. The kinetic energy of 'M' slugs moving at a velocity of V ft/second is equal to  $\frac{1}{2} MV^2$  ft lb. It can be seen the same thrust and momentum can be achieved from either one slug given 10 ft/second acceleration or, ten slugs an acceleration of one ft/second. Using the above formula ( $\frac{1}{2} MV^2$ ), it shows:

A) One slug given 10 ft/second =  $\frac{1}{2} \times 1 \times 10^2 = 50$  ft lb.

B) Ten slugs given 1 ft/second =  $\frac{1}{2} \times 10 \times 1^2 = 5$  ft lb.

Or, if we prefer to use the metric formula (in kilojoules) then it becomes,  $\frac{1}{2} MV^2 = kJ$

where M = mass, Kg

V = velocity, m/s.

Therefore, 'B' is more efficient because it wastes less energy and produces the same momentum by accelerating a large mass of air at a low velocity.

The propeller's axial momentum theory differs from the blade element theory, which deals with the forces acting on the propeller, the lift, and drag characteristics.

See **Rankine, W. J. M; Propeller aerodynamics; Propwash-thrust.**

**Axial thrust** This is another term for the propulsive thrust produced by the propeller.

**Axis of constant moment** This is an alternative name for the aerodynamic centre, the point about which the wing's pitching moment is zero with changing angle of attack.

*See* **Aerodynamic centre (a. c.).**

**Axis of rotation** It refers to the axis of propeller or helicopter rotor blade rotation. It is located at right angles to the plane of rotation of the propeller axis or rotor blades. A helicopter's rotor blades rotate around an imaginary line perpendicular to the rotor blades tip path. The axis is fixed relative to the longitudinal axis on an airplane but can be tilted in a rotor blade system to provide directional control of the helicopter.

*See* **Authorotative thrust or force; Rotor force.**



## B

**Backside of the power (drag) curve** The backside of the power curve can be seen on *Diagram 60, Thrust & THP Required & Available Curves*.

Moving to the left of the diagram from  $V_x$  minimum power on the curve, the true air speed in Knots decreases and the power required increases. Therefore, the aircraft will require more power to fly at a slower speed and maintain altitude due to the increase of induced drag with increasing angle of attack.

In an extreme situation, applying up elevator will result in the aircraft descending because there is no longer sufficient power to maintain straight and level flight. When more power is required as the speed decreases, the airplane is said to be speed-unstable.

See **Region of normal and reversed command; Diagram 60, Thrust & THP Required & Available Curves**.

**Backwash** The term backwash is an alternate name for slipstream, which is the total airflow moving over the aircraft due to forward flight. The term is now seldom used.

See **Slipstream velocity**.

**Balance** The airplane's state of equilibrium when all four forces of (lift/weight and thrust/drag) are in balance. It also refers to a control surface's mass balance.

See *Diagram 33, Forces in cruise flight – airplane; Lift/weight equilibrium*.

**Balance area** That portion of a control surface's area located ahead of the hinge line axis as found on a horn type balanced control surface. Its purpose is to relieve some of the pressure required by the pilot to move the control surface.

**Balance tab** Balance tabs are used to relieve control stick force on the primary control surface such as a fixed tailplane with an elevator.

Balance tabs move automatically in the opposite direction to the primary control surface to which they are attached. A mechanical linkage will move the balance tab down whenever the pilot moves the primary control surface upwards and vice versa.

Pilot input movement to the primary control surface is transferred to the balance tab by means of a rod connection. The balance tab is attached to and works in opposition to the primary control surface's trailing edge by providing a small aerodynamic reaction in the appropriate direction. The airflow acting on the tab assists the primary control surface to move by reducing the hinge moment. The action is similar to the servo tab.

Compare with the anti-balance tab, which works in the opposite sense.

It is also known as a trim balance tab.

*See* **Spring tab; Flettner tabs; Trimming devices; Geared tabs.**

**Balanced control surface** Aerodynamic or mass balancing may be used to balance a primary control surface.

An aerodynamic balance is one with an inset hinge or horn balance. They provide increased control feel and ease of movement (less stick force per g) for the pilot. The aerodynamic balance moves the centre of gravity of the control surface closer to its hinge line, which reduces the hinge moment force required from the pilot to move it.

A mass balance is used on some aircraft to damp out any control surface flutter.

Although it did not fly, the Frenchman, Felix Du Temple (1823–1890), designed the first aircraft with a balanced rudder in 1857.

*See* **Tabs; Stick force per 'g'.**

**Balloon** Flap extension in flight, if too rapid can cause the aircraft to rise above the flight path, or balloon. When flaps are deployed, the angle of attack must be reduced to counter the extra lift produced by the extending flaps. Ballooning is more common on low wing types than on high wing aircraft, which tend to do the opposite and to initially ‘mush’ downwards.

**Barn door flaps** The term ‘barn door flaps’ is a colloquial expression for relatively large-area flaps. Double-slotted, triple-slotted and Fowler flaps, etc, qualify as barn door flaps.

**Basic stall** A basic stall is also known as a straight-ahead stall or a 1g stall.

**Belly fairing** Many modern jet transports, Bizjets, and other aircraft have a very distinctive bulge on the underside of the fuselage. This is an aerodynamic refinement to reduce the drag at the fuselage/wing root attachment point, particularly on aircraft with pressurized fuselages.

For relatively slow aircraft, the fairing can be quite short and blunt; however, for faster aircraft the belly fairing is required to be much longer to reduce profile drag. The airflow passing over the fairing ‘sees’ the wing as being mounted higher on the fuselage, which is a more aerodynamically efficient location than either a low or high wing position.

It also has the advantage of providing extra storage space for the undercarriage, fuel or as a baggage pack.

It is also known as ventral fairing.

**Bernoulli, D.** The first man to have any influence on the subject of aerodynamics was the well-known Daniel Bernoulli (1700–1782). He was a Swiss national born in Holland, from a family very well educated in many and varied fields. In conjunction with his father Johan Bernoulli, he researched fluid dynamics at the University



**Photo:** A streamlined belly fairing helps smooth the airflow over the wing/fuselage junction below the fuselage. Belly fairings are now a common feature on jet transports as can be seen in this photo of a Boeing 777-300ER.

in Basle, Switzerland, where in 1738; he introduced his theorem, which he published in his book *Hydrodynamics*. His theorem relates the pressure and velocity in an incompressible, inviscid flow and is best remembered by pilots and aerodynamicists as the Bernoulli theorem and the related equation ( $p + \frac{1}{2} \rho V^2 = \text{constant}$ ). However, his equation is actually a misnomer. The majority credit for this famous theorem is shared with a lesser-known scientist, Leonhard Euler (1707–1783) an associate of Bernoulli who was also a Swiss national working in Basle. The theorem introduces the venturi effect, which we can now relate to the pressure distribution over the wing, where the pressure decreases with an increase in speed producing lift across the wing's upper surface.

**Bernoulli's theorem** Bernoulli's theorem states 'the dynamic pressure (pressure energy) plus the static pressure (kinetic energy) equals a constant', shown by the equation ( $p + \frac{1}{2} \rho V^2 = \text{constant}$ ).

A constant mass, inviscid, incompressible fluid flow traversing a tube of uniform diameter will have a constant velocity and dynamic pressure. On reaching a restriction in that tube, a venturi, the mass flow will experience an increase in velocity and dynamic pressure. This will result in a drop in the static pressure, due to the dynamic pressure ( $\frac{1}{2} \rho V^2$ ) increasing and the static pressure decreasing, the total pressure remains constant. This is in accordance with the conservation of energy; it also obeys Newton's laws of motion. The dynamic pressure (pressure energy) plus the static pressure (kinetic energy) equals a constant; that is, the total pressure remains the same. All aerodynamic forces are directly proportional to dynamic pressure.

Bernoulli's theorem explains the working of a piston-engine's carburetor, an aircraft's wing, and many other applications. The carburetor has a venturi situated within the barrel; air is drawn in through the intake and the pressure decreases in the venturi, sucking in vaporized fuel for delivery to the engine's cylinders.

The aircraft's wing works in a similar fashion to the venturi. In accordance with Bernoulli's theorem, air flowing over the upper contoured surface of the wing (which has a longer path length than the lower surface) increases in velocity due to a decrease in the pressure gradient resulting in aerodynamic lift. The purist may argue a wing is not half a venturi, and inverted flight, ground effect and the influence of the angle of attack cannot be explained with this theory. However, if the reader understands the working of the venturi tube then the principle applied to the wing will be obvious. This theory is commonly used in basic aerodynamic textbooks for its simplicity. However, the theory breaks down when used mathematically, and should only be used as a theoretical example. The Bernoulli theorem does not take into account the airflow underneath the wing, which also contributes towards the lift.

Four types of energy in the form of heat, pressure, kinetic and potential energy can be found in a streamlined incompressible flow. Below 200 KTAS/Mach 0.3 and 10000 feet, heat and pressure energy can be ignored. This leaves the equation where potential energy plus kinetic energy equals a constant. Using this equation in the lift and drag formulas it becomes  $\frac{1}{2} \rho V^2$  where the symbol  $\rho$  (rho) represents the air density. The equation becomes static plus dynamic pressure equals a constant.

It should be noted here, Bernoulli was interested in water pressure when working on his famous equation – and this was in the days long before aviation became a reality. Because Bernoulli's equation assumes constant density, it is not applicable to compressible flows above 200 KTAS/Mach 0.3.

**Best range speed** The best range speed is defined as the speed at which, 'the least amount of fuel used per unit distance' in nautical miles. It is normally at 1.3 times the  $V_y$  minimum drag speed.

*See Piston-engine power curves – airplane.*

**Beta angle** A propeller's beta angle is a negative angle between the propeller blade and the plane of rotation. The propeller is operating at its beta angle during reverse thrust operations.

**Beta mode** The propeller is in the beta mode when reverse thrust is applied.

**Beta range** The propeller's beta range extends from the fine pitch position through a blade angle of zero degrees to the reverse pitch position; it is used for reducing the aircraft's speed after landing.

Using the propeller in this range is known as the beta mode.

**Betz, A.** Professor Albert Betz (1885–1968) was a German aerodynamicist at the Gottingen Aerodynamic Research Institute who discovered the advantage of sweptback wings to delay the effects of high-speed compressibility. He also worked in conjunction with Professor Ludwig Prandtl (1875–1953) and Maxwell Munk (1890–1986) in 1932, on research into pressure distribution, span loading, wingtip vortices, and their effect on induced drag.

**Bifurcated shock wave** An alternate name for Lambda shock wave.

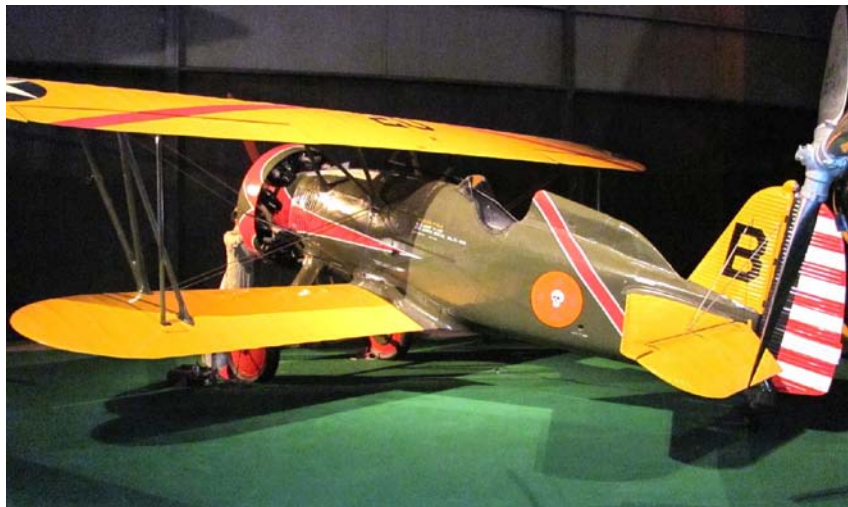
**Biot-Savart law** The original Biot-Savart Law related magnetic fields to electrical currents and was first developed about 1820. The law later became adopted into aerodynamic terminology, where it is used to calculate the velocity induced by vortex lines, which relates the velocity induced by a vortex filament to its strength and orientation. Compared to the original application, the vorticity and current are reversed.

The original Biot-Savart Law was devised jointly by the Frenchman Jean-Baptiste Biot (1774–1862) an astronomer, physicist and mathematician and his co-worker Félix Savart (1791–1841) a professor, in 1836 at the College de France. Together, they worked on the theory of magnetism and electrical currents.

**Biplane** A biplane is an aircraft (airplane or glider) with two main wings superimposed one above the other. The wings are generally staggered, most commonly with the top wing forward of the lower wing (forward, or positive stagger), for improved performance. It is rare to find rearward (or negative stagger) stagger; the Beech Model 17 Staggerwing is an exception. A biplane without stagger is also rare, and is known as an orthogonal biplane; the 1928 Vickers Vildebeest torpedo-bomber is one example.

A biplane with forward stagger will be set at a greater angle of incidence to induce the upper wing to stall before the lower wing;

the total lift is greater on the upper wing as opposed to a rearward staggered wing where the lift is greater on the lower wing. The lower wing usually has less span to provide greater wing tip ground clearance when taxiing. Dihedral is normally applied to the lower wing unless both wings have it. Dihedral also increases ground clearance on the lower wing. Another common feature is for the upper wing to be swept back a few degrees to increase yaw-roll coupling for increased manoeuvrability, and the lower wing to be straight.



**Photo:** The Boeing P-12E fighter was one of the USA's most successful biplane fighters during the 1930s.

Several early pioneers of flight prior to 1900 used biplane gliders and airplanes. The word biplane originated in 1874, but the meaning as we know it today, dates from 1908. The biplane configuration lost appeal to the superior monoplane construction during the 1920s–1930s, with only a few biplane types built over the following years, for example the Pitts Special, mainly for the biplane pilot/enthusiast.



Professor Albert Betz (1885–1968) and later Dr. Maxwell Munk (1890–1986) at NACA Langley performed considerable aerodynamic research on biplanes.

*See* **Orthogonal biplane; Stagger.**

**Biplane cellule** This term refers to the biplane's wire braced wings, which means they both work together to produce lift.

**Biplane drag** A biplane with a wing area equivalent to that of a monoplane, will have half the induced drag of a monoplane, because induced drag is proportional to the square root of the span loading. The lower wing produces more induced drag than the upper wing, due to the lower wing's lift vector being tilted rearwards more than the upper wing. Therefore, the lower wing has a reduced lift/drag ratio with more induced drag than the upper wing. Although the induced drag is less than a monoplane's wing, the biplane's struts and wire bracing add extra form drag and the wing is inherently less aerodynamically refined than a modern monoplane's wing. Therefore, for a comparable size, the biplane's wing suffers from increased drag and is less efficient.

**Biplane efficiency factor** The biplane's wings efficiency factor is determined by the values of its gap/chord ratio, decalage, wing thickness and tip overhang.

**Biplane equation** *See* **Prandtl-Glauert interference factor.**

**Biplane interference** The increased air pressure field below the upper wing of a biplane interferes with the reduced air pressure field generated by the lower wing. The lift coefficient of each wing is affected by the other wing. Therefore, the lower wing is less efficient than the top wing due to the interference and downwash caused by the wing above. The lift vector component on the lower wing is angled further rearwards, reducing the lift/drag ratio of the

lower wing. It follows, the lower wing produces more induced drag than the upper wing. Staggering the wings by moving them out of alignment helps to reduce this problem and improve performance. The maximum lift coefficient ( $C_L$  max) for a biplane is about 95% of an equivalent monoplane, depending on the wing's gap and stagger.

See **Stagger**; **Negative stagger**; **Orthogonal biplane**.

**Blade** The term blade refers to an aerodynamically shaped airfoil that rotates about an axis. Examples are, a propeller, helicopter rotor blade or an axial compressor blade as found in a gas turbine engine.

**Blade angle ( $\beta$ )** A Propeller or helicopter's blade angle  $\beta$  (beta) is akin to an airfoil's angle of attack. The blade angle is the variable angle between the rotor blade's chord line and the plane of rotation. The pilot via the collective controls the blade angle.

See **Pitch angle**; **Angle of attack ( $\alpha$ )**.

**Blade area** The blade area is the total area of all the propeller's blades or the helicopter's rotor blades.

**Blade back** The propeller blade's back corresponds to the upper surface of an aircraft's wing. With a tractor prop, the back of the prop is seen when viewed from in front of the aircraft!

**Blade cuffs** Propeller blade cuffs increase the area of the blade root, thus increasing the propeller efficiency by 1–3% and reducing the propeller noise by approximately three dB.

Also known as extended propeller blade roots.



**Photo:** The Lockheed Orion maritime patrol aircraft has blade cuffs on each of its propellers.

**Blade element** A thin section of propeller blade corresponding to an airfoil section, perpendicular to the blade's major axis, is known as a blade element.

See *Diagram 68, Propeller Pitch*.

**Blade element theory** The blade element theory is the most widely accepted theory on how propellers work, along with parts of the axial momentum theory. The blade element theory is similar to the wing's airfoil section theory.

The blade element theory is associated with the forces of lift and drag characteristics acting on the propeller (or helicopter rotor blades) as it moves through the air at a uniform velocity. It also includes the blade's shape and number of blades and assumes

the propeller blade to be made from an infinite number of blade elements, hence the name blade element theory.

The blade element theory was first introduced by William Froude (1810–79) an engineer and naval architect in 1878, when he was working on ship's propeller theories. Stefan Drzewiecki (1844–?) further developed the blade element theory from 1892 to 1920 and he has been credited with the majority of the research work. Herman Glauert (1892–1934) is credited with adapting the blade element theory to aircraft propellers in 1935.

It is also known as the momentum vortex blade element theory.

*See* **Propeller aerodynamics**.

**Blade element momentum theory (BEMT)** Combining the blade element theory based on Newtonian physics and the momentum theory introduced by William Froude (1810–1879) is a convenient method to describe propeller and rotor blade theory of lift/thrust.

*See* **Blade element theory; Momentum theory**.

**Blade face** The relatively flat propeller surface corresponding to the wing's lower surface is known as the blade face. It is the side of the blade as seen from the rear of a tractor prop.

It is also known as the thrust face.

**Blade flapping** A helicopter's main rotor blades will flap up and down during their rotation, flapping about the horizontal-flapping hinge. The rise and fall of the blades is due to the inertia and centrifugal forces of the blades and changes in lift force as the blades rotate.

It is also known as flapping.

**Blade loading** The blade loading on a propeller is equal to the engine's brake horsepower (BHP) divided by the total propeller blade area.

Blade loading on a helicopter's rotor system is equal to the all up weight of the craft divided by the total blade area expressed as pounds/square feet or  $\text{kg/m}^2$ .

See **Propeller blade loading**.

**Blade pitch angle (BPA) & twist** The blade angle is defined as the angle between the helicopter's rotor or propeller's plane of rotation (A-B on *Diagram 68, Propeller Pitch*) and the propeller blade's chord line (A-D) combining the helix angle and the angle of attack.

Each blade element travels on a different helical path and to be aerodynamically efficient must meet the relative airflow at a small angle of attack of  $3-4^\circ$ . To achieve this constant angle of attack along the length of the blade the propeller must therefore be twisted. This is known as the propeller's geometric twist where the angle between the blade chord and its plane of rotation varies along the span. This requires the blade angle to be greater at the root with a gradual reduction towards the tip, (washout) as mentioned above. The geometric pitch of the propeller then remains constant ( $P = 2 \pi R \tan \theta$ ) due to the blade angle decreasing as the blade radius increases. The varying blade angle is essential to maintain the same angle of attack along the blade, to ensure the full length of the propeller blade performs at the maximum lift/drag ratio (L/D). The actual blade twist is designed to provide the correct angle of attack at the design cruise speed.

Although the blade twist is associated with the geometric pitch, it must not be confused with the definition of blade angle and pitch. The blade angle is measured in degrees between vectors AB and AD, while the pitch is measured as a length in inches (B-D) on *Diagram 68, Propeller Pitch*.

The term blade pitch angle also applies to a helicopter's rotor blades.

See *Diagram 68, Propeller Pitch*.

**Blade sailing** Blade sailing is a phenomenon associated with helicopter operations during engine start-up and shutdown procedures. Two-blade rotor systems are more prone to the problem than three or more blade systems.

The problem occurs when the rotor RPM is low and the centrifugal force acting on the blades allows them to freely flap-up or down excessively, especially in strong, gusty wind conditions. The rotor blade moving into wind experiences an increase in speed compared to the retreating blade, which flaps-up as opposed to the retreating blade, which does the opposite and flaps-down, possibly striking the tail-boom in the process.

**Blade section** The blade section is the shape of the propeller blade element. It is akin to the airfoil section.

**Blade twist** A propeller's natural twist, known as blade twist, reduces the blade angle from root to tip. It is the propeller's counterpart of wing washout.

**Blended wing/body** A blended wing/body is an aircraft containing a streamlined, merging intersection of the wing root and fuselage. It is similar to a large-scale fillet, but where a fillet is an added extra to the airframe, the blended body intersection is a continuous part of the aircraft structure. The blended wing/body is a feature of many modern high-speed jet fighter aircraft with the General Dynamics F-16 Fighting Falcon and the Lockheed SR-71A Blackbird, both being very good examples.

Due to a high lift/drag ratio, it produces relatively less drag making it more fuel efficient, the lift at high angles of attack is increased and a weight saving is achieved through a lighter and more rigid structure.



**Photo:** The Rockwell B-1B Lancer with a blended wing/body on display at the Museum of the US Air Force in Dayton, Ohio.

**Blended wingtips** A blended wingtip consists of a winglet blended into the wingtip area to reduce the interference drag at their junction. A sharp angle between the two airfoils causes a stronger trailing vortex with greater induced drag, which is what the winglet is intended to reduce. Therefore, the blend reduces the vortex intensity and hence, reduces interference and induced drag. The streamlined blend is also more aesthetically pleasing.

They are used on several types of aircraft from sailplanes to Bizjets and Boeing airliners.

*See Winglets; Wingtip devices.*



**Photo:** The inboard side of a blended wingtip on the port wing of a Boeing 767-300, as seen from the passenger cabin in flight.

**Blowback** The blowback or closing of an airplane's spoilers in flight is due to the dynamic air pressure closing them against the force of the actuating device.

Blowback of a helicopter's rotor disc can occur when a helicopter moves forward in flight. It is caused by gyroscopic precession of the rotor disc causing the front of the disc to rise.

*See **Flap-back.***



**Blown flap** A blown flap is a form of boundary layer control system.

A high velocity stream of air is blown across the upper surface of the deflected flaps to maintain full effectiveness. The purpose is to maintain an attached boundary layer over the deflected flaps to maintain continued effectiveness and allow lower approach speeds for landing.

The aircraft's jet engine(s) supplies compressed air directed to follow the upper surface contours of the flaps (due to the Coanda effect) to considerably energize the boundary layer airflow and increase their lift. The increased airflow velocity over the flaps has the same effect as if the aircraft itself were flying faster. The advantage is lower take-off and landing speeds and smaller, lightweight flaps.

Blown flaps are of two types. The first type is the externally blown flap, where the slotted flaps redirect the jet engine's exhaust downward to follow the flap contours due to the Coanda effect. The lift coefficient is thus greatly increased improving low-speed lift. The second type of blown flap again uses the Coanda effect to direct increased airflow over the upper surface of the wing and flaps from engines mounted above the wings.

The British built Hunting H.126 research aircraft was initially used to flight test the blown flap system, achieving a very low stall speed of 28 Knots (32 MPH). The H.126 was initially flight tested at RAe Bedford, UK, before being shipped to the USA for wind tunnel testing in California. It made its first flight in March 1963.

Blown flaps were first used operationally on the Lockheed F-104 Starfighter; they are known on this plane as Attinello flaps after their designer, John Attinello. Blown flaps were common on a few jet fighter aircraft during the 1950–60s, after development by NASA. The BAC TSR.2 (Tactical Strike and Reconnaissance) jet bomber was also designed with powerful blown flaps mounted on its small delta wing. The wing area was 65.03 square meters (700 square feet) with high wing loading, which would have required a high landing speed had it not been for the blown flaps. The



**Photos:** Two views of the Hunting H.126 research aircraft used to flight test the blown flap system. The H.126 shown here is on display at the RAF Museum in Cosford, England.

supersonic TSR.2 first flew on 27 September 1964, but the British government the following year canceled the project.

However, due to the complexity of the blown flap system and frequent maintenance requirements, the idea faded from favor for many years. After revised testing in the 1970s, the idea of blown flaps was employed on the Ukraine built STOL, civilian/military, twinjet Antonov AN-80, and the Boeing (MDD) C-17A Globemaster III, a military transport. Most jet transport aircraft do not have continuous flaps along the wing's trailing edge but have an open space to allow the jet engine's exhaust a free, uninterrupted passage. The Airbus A-380 jet transport, has blown flaps but does not have any gap – the flaps are continuous along the wing's trailing edge; the four jet engines direct their exhaust onto the flap's under surface to increase their lift for the approach and landing.

The Americans use the term blown flap as apposed to the English term of jet flaps.

**Blunt nose re-entry vehicle** A sharp pointed nose is ideal for supersonic aircraft; however, for the hypersonic speeds of re-entry vehicles (space capsules and the Space Shuttle for example) a blunt nose is an essential requirement for heat dissipation.

The blunt nose cone causes a strong bow shockwave to form, which requires considerable energy from the airflow to move it ahead of the nose cone. The airflow through the shockwave experiences a decrease in velocity, while the density, pressure and in particular, the temperature, all increase due to compression, converting the kinetic energy to thermal energy. Heat is transferred to the vehicle surface via radiation, convection, and conduction. The area between the shockwave and the nose cone diverts some of the heat away from the body of the re-entry vehicle and into the passing airflow.

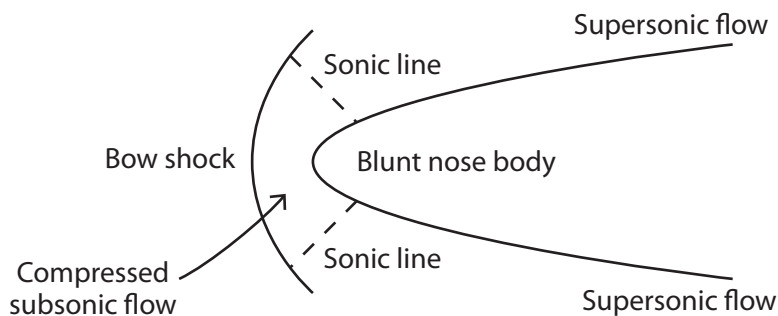
A laminar boundary layer flow within the bow shockwave cone is preferable to a turbulent flow, which has the disadvantage of faster moving air particles causing a greater rate of undesirable heat transfer to the vehicle's body surface.



**Photo:** The North American X-15 rocket plane has a blunt nose to form a bow shock wave during supersonic/hypersonic flight. This aircraft is located at the National Museum of the USAF, in Dayton, Ohio.

Working at the NACA Ames Aeronautical Research Centre in 1951, the engineer H. Julian Allen (1910–1977), introduced the idea of a blunt nose for re-entry space capsules. They were flight tested on the lifting body research aircraft such as the Martin X-24A, Northrop M2-F3, and the Northrop HL-10, and it later became a prominent feature on the Space Shuttles. The blunt nose theory was initially classified information when first discovered and was not disclosed to the world until 1958.

Alfred J. Eggers Jr. (1922–2006) was well known for his supersonic research and re-entry physics. He joined NACA Ames in 1944 and he assisted H. Julian Allen. **Diagram 8, Detached Normal Shockwave**, shows the detached bow shockwave, the compressed subsonic flow and the sonic lines; they represent the boundary between the supersonic flow and subsonic flow.



**Diagram 8, Detached Normal Shockwave**

See **Sonic boom; Refracted boom.**

**Body axes** By convention, the body axes are a right-handed orthogonal axis system, where each axis acts perpendicular to each other. They are fixed relative to the aircraft fuselage and usually but not necessarily, pass through the centre of gravity. The three axes are the X body axis, Y body axis and Z body axis, about which the aircraft rolls, pitches and yaws respectively. A force acting on the aircraft along an axis is known as a body-axis force.

- X body axis is aligned (longitudinally) along the fuselage
- Y body axis is transverse from wingtip to wingtip
- Z body axis is perpendicular to the Y and X axes.

G. H. Bryan introduced the body axis terms in 1903. The alternate name is body-fixed axes.

*See Aerodynamic axes.*

**Body drag** Body drag includes all adverse drag producing items including drag caused by the fuselage frontal area, engine nacelles, tailplane, air gaps, gear and struts, etc. Body drag is a sub-division of parasite drag.

*See Drag.*

**Body-fixed axes** *See Body axes.*

**Body lift** The lower surface of the fuselage of a supersonic aircraft can also contribute to the lift generated by the wings at a positive angle of attack. The additional lift is created by the fuselage wedging the airflow downwards thus creating an upward lift reaction.

It is also known as compression lift or wave riding.

**Boom carpet** An aircraft flying at supersonic speed produces shockwaves that reach the ground along a corridor below the aircraft's flight path. It is known as the boom carpet, due to its likeness to a 'red carpet' being unrolled.

**Boundary layer** The boundary layer is the thin region of retarded viscous airflow next to the surface of a body (aircraft's wing) in flight. Portions of the boundary layer may be either a laminar attached flow or turbulent detached flow (boundary-layer turbulence).

The effects of viscosity are found within the boundary layer, and present the characteristics due to skin friction drag and pressure drag. The air molecules nearest the surface of the body will be at, or almost at a standstill. When the molecules become stationary on the surface, it is known as the 'no-slip condition'. Successive layers (or lamina) of airflow above the airfoil surface will move progressively faster, due to the viscosity of the fluid. It is akin to a pack of cards thrown onto a table – the upper cards slide further than the lower cards. The boundary layer thickness (represented by the Greek letter delta  $\delta$ ) increases in depth until reaching a velocity of 99% of the free air stream velocity, which is considered to be the upper level of the boundary layer. The boundary layer thickness, known as the displacement thickness, depends on the Reynolds number of the flow.

The boundary layer (or flow attachment) characteristics change with chord distance from the leading edge, changing from a laminar flow to a turbulent flow. This is due to the shearing effect of the boundary layer and the increasing severity of the adverse pressure gradient as the flow progresses across the wing, with skin friction drag reducing as pressure drag increases across the upper surface. Shear stress increases in the greater turbulent layer due to unstable eddy formation, which creates increased boundary layer thickness and a greater increase in boundary layer velocity.

The boundary layer alters the effective profile of the wing by reducing the effective camber; therefore, the free air stream flowing above the boundary layer 'sees' a slightly different shape to the physical wing shape. The influence of the boundary layer is accounted for in the wing's lift coefficient, while the associated skin friction drag is allowed for in the drag coefficient. For high-speed flight, compressibility must also be taken into consideration.

The transition from laminar to turbulent flow in the boundary layer occurs readily at a significantly high Reynolds number ( $Re$ ) of around 500 000, in the adverse pressure gradient where the boundary layer thickens. The  $Re$  is relatively small in a thicker boundary layer with a viscous flow compared to a higher  $Re$  in a thinner inviscid boundary layer.

An increase in angle of attack will also produce a thickening of the boundary layer inducing transition to a turbulent flow. The boundary layer thickens due to friction drag heating close to the wing's surface with increasing velocity, which causes an increase in airflow viscosity and a decrease in air density. The boundary layer thickness varies with the square of the Mach number. However, the boundary layer thickness decreases with an increase in subsonic velocity because it is inversely proportional to the Reynolds number. Therefore, the boundary layer thickness depends mainly on the aircraft's velocity. The boundary layer becomes the wake in the downstream flow.

A laminar pressure gradient has:

- low thickness laminar flow boundary layer
- low skin friction drag, about one third that of the turbulent flow friction drag
- relatively zero speed at wing surface
- gradual increase in speed to top of the boundary layer.

A turbulent pressure gradient flow has:

- greater thickness turbulent boundary layer
- increased pressure drag and friction drag
- higher flow velocity next to surface
- increased kinetic energy
- velocity in boundary layer increasing faster to free stream flow.

Professor Ludwig Prandtl (1875–1953) established the idea of the boundary layer in 1904. Later, in 1908, Paul Blasius (1873–1970) a German fluid dynamics engineer and a student of Prandtl introduced his theorem – it became named after him as the



Blasius theorem, which described mathematically Prandtl's two-dimensional boundary layer drag on a flat plate. The boundary layer theory was further developed by NACA during the 1930s when they introduced the laminar flow wing, which was first used on the North American P51 Mustang of WW II fame. The boundary layer profile is also known as the Blasius flow profile.

*See Transition point/region; Blown flap; Laminar flow airfoils; Turbulent flow; Navier-Stokes Equations; Diagram 1, Adverse and Favourable Pressure Gradients.*

**Boundary layer control system** The purpose of a boundary layer control system is to increase the kinetic energy in the boundary layer, which will increase the maximum lift, reduce the drag, improve the airflow over the control surfaces and wing's upper surface to lower the stall speed.

The various methods employed are:

- vortex generators
- boundary layer suction
- blown boundary layer
- propeller slipstream or jet thrust over the wing.

Boundary layer suction is designed to suck air into the wing through holes strategically placed on the upper wing's surface. This helps to replace the boundary layer's low energy with airflow of increased velocity from outside the boundary layer.

The opposite of boundary layer suction is a blown system. Air taken from a jet engine's compressor is ducted to vents in the wing's upper surface and ejected or blown at high velocity into the boundary layer. It has the same effect and advantages as the suction system but is far more practical from an engineering point of view.

Boundary layer control is also known as supercirculation.

*See Vortex generators; Propeller slipstream.*

**Boundary layer drag** Boundary layer drag (profile drag) is the sum of the form drag and the skin friction drag. The boundary layer drag is due to surface friction, which induces form drag. Surface friction is reduced by smooth surfaces while streamlining reduces form drag. The friction drag causes the boundary layer to be retarded, energy is dissipated which destroys the laminar flow and the boundary layer thickens into a turbulent flow.

It is also known as the boundary layer separation drag.

See **Drag; Pressure drag; Vortex generators.**

**Bound vortex** The bound vortex is part of the circulation of airflow around an airplane's wing.

A vortex may be either a free vortex or a bound vortex. According to Helmholtz theorem, a bound vortex continues as two trailing wingtip free vortices. The bound vortex is that part of the horseshoe vortex that is 'bound' to the wing's lifting line; it is therefore also known as a line vortex. Trailing wingtip vortices (or wake vortices) continue to emanate from the bound vortex during flight.

See *Diagram 40, Horseshoe Vortex; Cusp; Lifting line theory; Helmholtz vortex theorem; Kutta condition.*

**Bow shock wave** An alternate name for an unattached shockwave, is bow shock wave, or simply bow wave. It forms attached to the front of an aircraft's nose or on the wing's leading edge as the aircraft accelerates through the low supersonic area and by Mach 2.0, the bow wave is fully established.

See **Shockwaves; Viscous flow.**

**Boyle-Mariotte Law** See **Boyle's Law.**

**Boyle's Law** Robert Boyle (1627–1691) was an English natural philosopher, chemist, physicist, and a founding member of the Royal Society. He published the first of the gas laws named after him in 1662, as follows:

- the volume of a gas decreases inversely proportional to its absolute pressure when the temperature remains constant. Where  $PV = K$  (constant), or written as  $P_1/V_1 = P_2/V_2$ . If the volume is doubled, the pressure is halved
- the volume increases proportionally to the absolute temperature
- the temperature will rise and the volume will decrease in proportion to the increase in absolute pressure
- if a gas is compressed or expanded with the temperature remaining constant, the compression or expansion is said to be isothermal
- Density is directly proportional to the absolute pressure.

The French physicist, Edme Mariotte (1620–1684) the father of French hydraulics, discovered the relationship between temperature, density, and pressure. However, in 1660 Robert Boyle earned the credit for the discovery.

Boyle's Law is also known as the Boyle-Mariotte Law.

See **General Gas Law; Charles' and Gay-Lussac's Law; Gay-Lussac's Law; Ideal gas law; Avogadro's law.**

**Brake horsepower** The power output of a piston-engine is found by coupling the engine to a test-bed dynamometer. This device measures the torque or turning force of the engine crankshaft in foot-pounds (not to be confused with pounds-feet of work). The torque in foot-pounds is converted mathematically into brake horsepower by using the following formula:

$$\text{BHP} = \frac{2 \pi T N}{33000}$$

Where BHP = 33000

$\pi$  = 3.1415... a constant

T = output torque

N = shaft speed in revs per minute.

One horsepower equals 550 ft-lb per second or 33000 ft-lb per minute.

The Scottish engineer, James Watt (1736–1819) introduced the term of horsepower.

*See Diagram 60, Thrust & THP Required & Available Curves.*

**Braking pitch** To provide maximum reverse thrust for braking, a negative propeller blade pitch angle (Beta range) is used.

*See Reverse thrust forces.*

**Brayton cycle** George Bailey Brayton (1830–1892) an American mechanical engineer introduced the Brayton cycle to describe the thermodynamic process in engines, based on the Carnot cycle introduced by the French physicist Nicolas Leonard Sadi Carnot (1796–1832) in 1823.

The Brayton cycle falls into the thermodynamic category, however, because it involves an aerodynamic airflow through the gas turbine (jet) engine, it has been included in this work, along with the associated definitions. George Brayton introduced the open or continuous cycle combustion process, which enabled the original development of the gas turbine engine. The change in airflow acting through the engine's four stages consists of two adiabatic and two isobaric, alternating changes all acting simultaneously. The pressure-volume (P-V) diagram familiar to all students of thermodynamics is shown here related to the gas turbine engine. *Diagram 9, P-V Diagram*, shows the gas turbine stages:

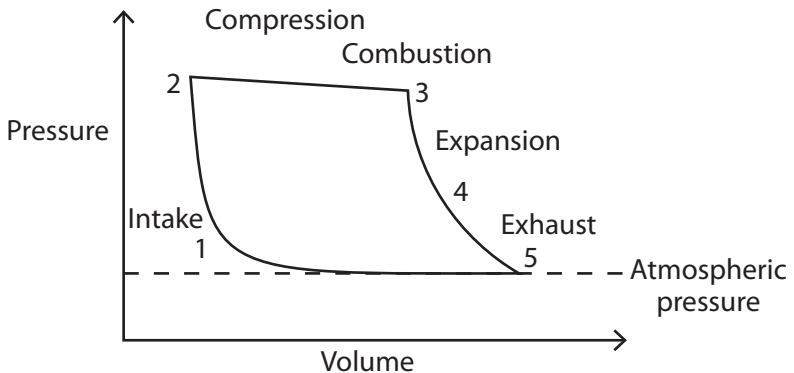
- 1–2 compression – adiabatic
- 2–3 combustion – isobaric
- 3–4 expansion – adiabatic
- 4–5 exhaust – isobaric.

Starting at point 1 and moving up to point 2, the first process involves the inducted airflow being compressed through the engine's compressor and thus adding pressure energy to the flow. The air velocity decreases and the pressure increases. This is an adiabatic or isentropic reversible process.

From point 2 to 3, heat energy is added (fuel burn) in the combustion chamber. The airflow remains at almost constant pressure or decreases slightly but the volume increases steadily due to the cross sectional area of the combustion chamber. This is the first isobaric process through the engine.

The third process of expansion from 3 to 4 shows the combustion gases on passing through the turbine section cause an increase in airflow velocity and a corresponding decrease in temperature and

pressure (Charles' Law). This is the second adiabatic or isentropic process.



**Diagram 9, P-V Diagram**

The fourth process of exhaust from 4 to 5 shows the temperature is reduced through the exhaust nozzle. Velocity remains constant through the jet pipe and rises slightly through the propelling nozzle, while the pressure remains constant through the jet pipe and decreases through the propelling nozzle. This is the second isobaric process.

The Brayton cycle is also known as the open cycle, because of the continues cycle of events occurring simultaneously through the engine. It is also known as the Joule's cycle, gas turbine cycle or thermodynamic cycle. Further, it falls into the category of internal aerodynamics.

**See Thermodynamics; Isentropic flow; Adiabatic process; Isobaric; Open cycle; Entropy; First Law of Thermodynamics; Second Law of Thermodynamics.**

**Breakaway thrust** Due to friction between the aircraft wheels and the ground, greater thrust is required to start the aircraft moving from rest than is required to keep the aircraft taxiing at a constant speed. It is also known as breakaway friction or starting friction.

**Breguet, L.** Louis Charles Breguet (1880 – 1955) was born in Paris, France to wealthy parents. He became a pioneer aviator and along with his brother, he built a helicopter in 1907. However, like many early helicopters, it was not successful being too unstable for safe flight. He formed his own company in 1900 at Douai, France and was a pioneer in the use of steel tubing for aircraft structures. He designed and built the Breguet 14, a reconnaissance/light bomber during WW I and many more successful designs followed over the following decades. The Airline he formed was the forerunner of Air France, still in operation today. He is remembered for the aircraft he designed and built and the Breguet range equation he devised.

**Breguet range equation** The Breguet range equation is used to determine the aircraft's safe cruise range, the still-air range, or the gross still-air range. The equation proves that the range will increase for a given fuel weight when the ratio of  $C_L/C_D$  is at a maximum. It involves the aircraft's lift/drag ratio, cruise speed, specific fuel consumption and the log of the ratio of the aircraft's given weights depending on the choice of range required as listed below:

- the safe cruise range is calculated from the fuel weight at the top-of-climb to the fuel weight at top-of-descent ( $W_1/W_2$ )
- the still-air range is calculated from the top-of-climb fuel weight to zero fuel weight
- the gross still-air range is calculated from the maximum all-up-weight to zero fuel weight.

The formula is as follows:

$$\text{Range} = V \times 1/\text{sfc} \times L/D = \text{Log } e (W_1/W_2)$$

Where  $V$  = speed

sfc = specific fuel consumption

$L/D$  = lift drag ratio

$W_1$  = weight at top-of-climb

$W_2$  = weight at top-of-descent.

The Frenchman Louis Charles Breguet (1880–1955) devised the Breguet range equation.

**Buffet** At high angles of attack, boundary layer separation ideally occurs near to the wing root's trailing edge. The resulting turbulent flow passes over the tailplane causing buffeting. This is an early warning symptom of an impending stall.

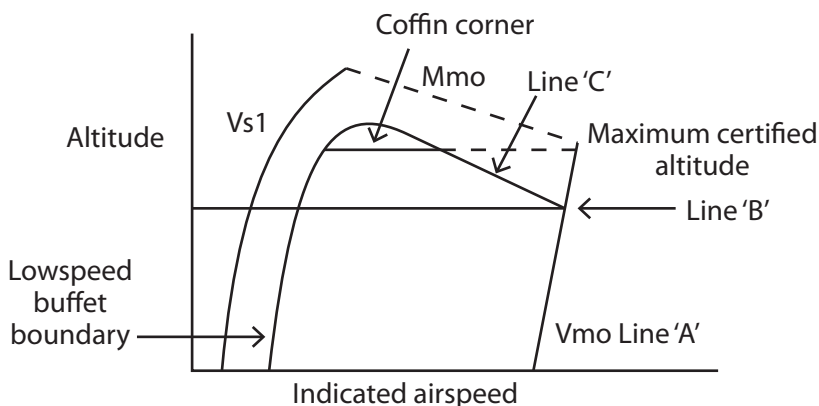
The aircraft's tail surfaces are a major cause of buffet at transonic speeds. This is the reason why some supersonic airplanes have delta wings and no tailplane.

An alternate name is judder.



**Buffet boundary** The buffet boundary applies to jet transport aircraft.

For a given airplane type, the buffet boundary depends on the maximum certified altitude and the maximum indicated air speed or Mach number. Low and high-speed buffet boundaries define the limits of the normal operating envelope for jet transport aircraft at high altitudes. The low-speed buffet boundary starts just above the stall speed when burbling occurs. Decreasing speed below the low speed buffet boundary may stall the aircraft.



**Diagram 10, Buffet Boundary**

**Diagram 10, Buffet Boundary**, shows the normal operating envelope where:

- the  $V_{MO}$  (line 'A') increases with altitude above the IAS/Mach number changeover speed, (line 'B') up to the maximum certified altitude
- the  $M_{MO}$  decreases with altitude. Coupled with the increasing IAS with altitude, the speed range between the stall speed and  $M_{MO}$  decreases with altitude and weight
- the high-speed buffet boundary (line 'C') starts just below the maximum Mach operating speed ( $M_{MO}$ ) at a speed where

buffeting – a high frequency vibration – starts due to flow separation caused by the shockwaves. The aircraft may lose control due to compressibility problems at high subsonic speeds

- the  $V_{S1}$  and  $M_{MO}$  limits merge to within a few Knots of each other forming the so-called coffin corner where any slight variation in speed, either less or greater, will result in loss of control. Turbulence or manoeuvres can place the aircraft close to the buffet boundary or within the coffin corner; therefore, precise aircraft control is vital at any speed near the buffet boundary limits. The maximum certified altitude must not be exceeded at any time!

See **Coffin corner**.

**Buffet corner** The buffet corner is a reference to the area at the top right-hand corner of the V-n manoeuvre envelope.

The junction of the maximum limit load factor (the top line) and the  $V_{NE}/V_{MO}$  maximum allowable indicated air speed (right-hand side limit line) on the manoeuvre envelope, defines the boundary of the buffet corner. These limits can also be seen on **Diagram 39, Gust Envelope**, from where the  $V_C$  line intersects the +50 FPS gust line and slopes down to where the  $V_D$  line joins the +25 FPS gust line.

The envelope corner is angled off to ensure the maximum allowable load factor reduces with increasing indicated air speed. The limit line denotes flight is not permissible beyond that area defined by this region of the envelope. This is due to the possibility of encountering wind gusts beyond allowable limits and due to high air speed causing load factors to exceed their limits, which could overstress the aircraft.

See **Manoeuvre envelope; Gust envelope**.

**Bullet** A bullet is a type of fairing located at the junction of an aircraft's cruciform or T-tail. The bullet is designed to prevent airflow separation (interference drag) due to conflicting airflow interference on the fin/rudder and elevator.

Bullet fairings were common on the early T-tail type aircraft, for example the Vickers VC-10, Illusion IL-62 (looks similar to the VC-10), Hawker Siddeley DH 121 trident and the Learjet 23. On most later aircraft, the bullet is missing with the horizontal tail being mounted slightly below the top of the vertical tail. However, there are a few exceptions, the Pilatus PC-12 is one example.



**Photo:** A Hawker Siddeley Trident 2 has a very prominent bullet fairing on its T-tail.

**Burble/burbling** The burble occurs at the critical angle of attack when the airflow no longer flows smoothly rearward over the upper surface of the wing. The airflow becomes turbulent or 'bubbles' near the trailing edge and the separation point moves forward, gradually spreading over most of the upper surface resulting in a loss of lift and an increase in drag. The low pressure on the upper surface increases to almost ambient pressure thus reducing lift. The drag force rises rapidly decreasing the speed even more so. The term is also applied to the high Mach numbers where the airflow separates due to the formation of shockwaves on the wings.

Burbling is the American's preferred terminology for flow separation.

See **Separation point & flow; Buffet; Stagnation point/line; Nibble.**

**Büsemann, A. Dr.** Doctor Adolph Büsemann (1901–1986) was born in Lubeck, Germany. In 1924, he earned his engineering degree and became a very prominent aerodynamicist. Using one of the early supersonic wind tunnels at the Kaiser Wilhelm Institute in the early 1930s, he specialized in high-speed aerodynamics and developed his idea of using sweptback wings for high-speed flight. Germany's (and the world's) first operational jet fighter was the Messerschmitt ME 262 twin-jet fighter with moderate swept wings followed by the Messerschmitt Me-163B Komet rocket powered interceptor, which became operational in 1944–5.

After the war, Dr. Büsemann worked for the Royal Aircraft Establishment at Farnborough, England for two years before moving to the USA in 1946 where he worked for NACA and later at the University of Colorado in Boulder. He was a world authority on high-speed flight and recognized as the originator of sweptback wings. However, it should be remembered that the first person to design a swept wing aircraft was John William Dunne (1880–



**Photo:** The Messerschmitt ME 262 Schwabe was designed by Dr. Büsemann. This Me 262 is on display at the National Museum of the USAF in Dayton, Ohio.

1949). The idea lay dormant for several years before Büsemann re-introduced the swept wing to the world.

*See* **Dunne, J. W.**

**Butterfly tail** The butterfly tail, also known as the V-tail consists of a two-surface empennage at a steep dihedral angle, as opposed to three surfaces as found on a normal tailplane. The two control surfaces work in the same direction (both up or down) as the elevator and in opposition (one up and one down) as the rudder. One advantage of the butterfly tail is the saving in weight with only two surfaces.

*See* **V-tail.**

**Buzz** The oscillation of the control surfaces, airplane surface skin, or other airframe structures at a high frequency.

The Lockheed P-80 Shooting Star fighter aircraft was the first aircraft recorded to experience buzz oscillations.

See **Flutter**.



**Photo:** A Lockheed P-80 Shooting Star USAF jet fighter in the National Museum of the USAF, in Dayton, Ohio.

# C

**Cadcam** The term cadcam stands for computer aided design/ computer-assisted manufacturing.

The Boeing 777 was the first airliner to be fully 100% cadcam designed, and 80% of the manufacturing process was performed by computer. The purist may say cadcam is not an aeronautical term. However, computers are used extensively by aerodynamicists and aircraft designers alike; therefore, it has a place in aerodynamic terminology. Cadcam followed on from the introduction of computational fluid dynamics.

The Boeing 777 made its first flight on 22 February 1987.



**Photo:** The Boeing 777 was the first airliner to be 100% cadcam designed.

**Calibrated air speed** Calibrated air speed (CAS) is the air speed indicator reading plus instrument error and position error; if allowing for compressibility, at International Standard Atmosphere (ISA) sea level conditions, the calibrated air speed (CAS) is equal to the true air speed (TAS).

CAS minus compressibility correction = EAS

EAS plus density error correction = TAS.

It is also known as rectified air speed, the British terminology.

See **Air speed; Compressibility correction.**

**Camber and camber line** The camber line is located midway between the wing's upper and lower surfaces; it is the airfoil's geometric centreline, and expressed as percentage camber. The airfoil's camber is the maximum distance between the mean camber line and the chord line. The camber therefore depends on the thickness/chord ratio, the maximum thickness point, plus other factors. A symmetrical airfoil has zero camber – the chord and camber lines coincide.

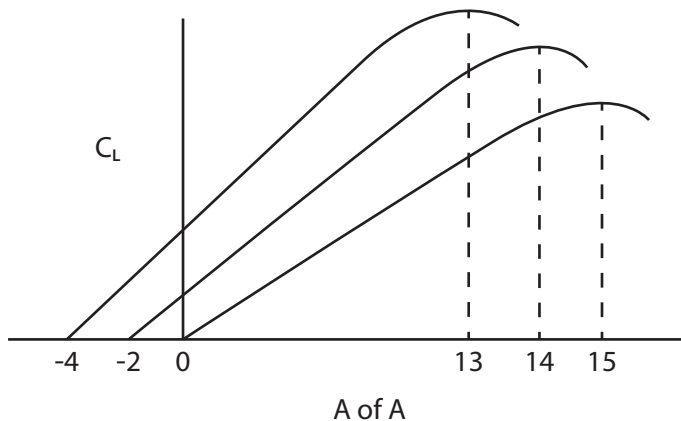
On a well-cambered airfoil, the camber increases up to around 25% MAC where the camber line arcs over the chord line. Greater camber produces lower air pressure above the wing and less drag at a higher lift coefficient and hence more lift at a relatively lower speed. If the camber is too far forward poor stall performance will result. With the maximum camber at 40–50% MAC, as found on laminar flow wings, the wing has a lower lift coefficient and less drag but is more suited to relatively higher speed aircraft.

Reference is also made to the upper and lower camber; this refers to the curvature of the upper and lower wing surfaces. A wing with positive camber has a greater convex upper surface than the lower surface. It is also known as thickness or thickness/chord ratio.

See **Diagram 4, Airfoil Terminology; Mean camber line; Maximum camber.**



**Cambered airfoil** Airfoils are either cambered or symmetrical. *Diagram 11, Camber & Lift Coefficient*, shows the symmetrical airfoil stalls at a higher angle of attack than a similar sized cambered airfoil; the maximum lift coefficient is less.



**Diagram 11, Camber & Lift Coefficient**

However, a cambered airfoil continues to produce lift, or thrust in the case of the propeller, below zero degrees angle of attack as indicated by a positive lift coefficient curve. When it is moving parallel to its zero lift line it produces zero lift, as the name implies. Lift is produced at these negative angles of attack due to the airfoil's camber creating a pressure differential between the upper and lower surfaces of the airfoil – lower pressure on the upper surface and increased pressure on the lower surface. A cambered airfoil with greater camber will have a lower stall angle of attack, and a greater maximum lift coefficient, compared to an airfoil with less camber.

In opposition to the cambered airfoil, when the symmetrical airfoil is at zero degrees angle of attack the lift coefficient is zero and therefore, it produces no lift.

The cambered airfoil has its aerodynamic centre located at 25% chord from the leading edge. Aircraft wings are mostly of

the cambered variety, apart from the vertical fin, which is always symmetrical. However, helicopter rotor blades may be of the symmetrical type.

*See Zero lift line.*

**Canard wings** A canard wing is a small lifting-surface type of stabilizer mounted on the forward fuselage – with or without elevators. The name is in fact, a misnomer; the word canard is French for duck, and being canards – they have their wings mounted towards the rear of their bodies, similar to the main wings of a canard-equipped airplane. On the other hand, does the name refer to a duck's bill? The English version of canard is a hyperbole (a gross exaggeration). However, in aviation terminology, canard is applied to the small forward wing on an aircraft.

In the true sense of the word, a canard is not a stabilizer surface. Although they do add to the stability of some aircraft, due to the wing loading of a canard being greater than the main wing for stability purposes. The canard surface produces an upload lift unlike the tailplane, which mostly produces a download. The canard's contribution to uplift allows the main wings to be made smaller and lighter, saving weight. The canard's uplift counteracts the nose-down tendency of the aircraft and enhances the aircraft's manoeuvrability and stability. Being forward of the main wings, the canard is always clear of the main wings downwash. It remains effective at high angles of attack and on some aircraft is designed to stall before the main wing due to greater wing loading, a smaller leading edge radius and a highly cambered upper surface, thus lowering the nose to prevent main wing stall. The canard also produces a trailing vortex flow over the main wing increasing the main wing lift over a greater speed range, in much the same way as a strake.

The canard also moves the main wing's centre of pressure forwards, the amount determined by the lift coefficients of the main and canard wings, their longitudinal distance apart, and the



**Photo:** The North American Aviation XB-70A Valkyrie has canard wings. The sole remaining Valkyrie is located in the National Museum of the United States Air Force in Dayton, Ohio.

relative wing areas of each wing. The aircraft's centre of gravity is located further forward, relative to a conventional plane.

The canard also has an influence on the rear of delta wing aircraft, which have short span wings with elevons for pitch and roll control on the trailing edge. They are devoid of flaps due to insufficient room along the trailing edge. The canard wing producing a nose-up pitch as it does, requires a forward stick movement to lower the elevons, which counteract the pitch-up. The elevons deflected downwards act in the same manner as deployed flaps, increasing the wing's camber to produce more lift and drag at high angles of attack. [Concorde does not have canard wings but uses fuel-transfer to shift the weight further aft than would normally be necessary, requiring down elevons, as mentioned above].

One disadvantage of canards is they are a hindrance on most jet transport aircraft when it parking at airports with an air bridge; the canards on the Russian Tupolev Tu 144, supersonic transport, retract into the side of the fuselage just aft of the cockpit, to clear the air bridge.

The Beachcraft Starship 1 twin-turboprop, executive aircraft, has canard wings, which swing back four degrees during cruise flight and swing forward during slow speed flight. The reason is to improve longitudinal stability over the aircraft's speed range.

Early aircraft commonly had canards (often referred to as tail first aircraft) but lost favor around the end of WW I, until their resurrection with the advent of the Swedish Saab Viggen jet fighter introduced in 1971. Canard wings are now popular on home-built light planes and some modern jet fighters.

**Canoe fairings** Canoe fairings are streamlined canoe shaped covers over the wing's flap tracks. They are commonly used on large jet transport type aircraft.

*See* **Flap track fairings.**

**Cantilevered wing** A cantilevered wing is one devoid of any struts or external wire bracing; all supporting structure is contained within the wing to reduce drag.

*See* **Struts, landing & flying wires.**

**Carson speed** The 'maximum lift/drag ratio speed times 1.32' is known as the Carson speed after its originator, Dr. Bernard H. 'Bud' Carson PhD (1934–2009), an American aeronautical engineer.

**Categories** The category in the context of weight and balance refers to normal, utility, or aerobatic categories. Most light aircraft fall into the first two groups. The normal category is defined as 'all normal manoeuvres with the angle of bank not exceeding 60°'. The utility category denotes limited aerobatic manoeuvres while the aerobatic category permits all normal and aerobatic manoeuvres.

*See* **Normal category; Limit & ultimate load factors; Ultimate load factor.**

**Cathedral** Cathedral (pronounced cat-hedral) is the term used when the aircraft's wings are drooped down (negative dihedral) towards the wingtips for stability purposes. The wings are more commonly known by the term anhedral.

*See* **Anhedral; Dihedral angle; Dihedral action/effect.**

**Cavitation** When the propeller tip speed approaches the speed of sound, a condition known as cavitation exists. Cavitation is due to the high tip speed creating a near vacuum on the suction face of each propeller blade near the tips, which reduces efficiency.

**Cayley, G, Sir** Sir George Cayley (1773–1857) was an English physicist, designer, far-sighted inventor, and Member of Parliament.

In 1796, Cayley's interest turned to theoretical and practical experiments on airfoils and in 1799, he discovered the basic aerodynamic forces acting on an airplane, defining the terms of lift, weight, thrust and drag.

Cayley's experiments also revealed the importance of the angle of attack, the effect of dihedral on lateral stability and discovered a cambered airfoil is more efficient than a flat plate. He is also noted for being the first to experiment in 1808, on the location of the wing's centre of pressure and its movement and how it affects the aircraft's longitudinal stability. Cayley put forward the theory that the wing's lift and engine thrust be treated as two entirely different

forces; that is, the lift equals the weight and thrust equals drag. He also stated the layout of an airplane should have a fuselage, tail with an elevator and rudder and the biplane or triplane wings fixed rigidly to the structure, not flapping, as found on ornithopters that were attempted to be flown in those early days. In 1799, Cayley sketched onto a silver disc a design for a separate rudder and elevator; the London Science Museum now holds the disc. Cayley's work became the basis for all aerodynamic studies to follow; he is the first true father of aerodynamics for heavier than air flight.

**Centre body cone inlet** The centre body cone inlet is positioned at the intake of a supersonic jet engine. A conical shockwave is induced by the centre body cone to reduce the speed of the intake flow by increasing the airflow pressure. A normal shock wave at the rear of the cone assists in reducing the airflow velocity further, from supersonic down to a subsonic flow before it enters the engine through its compressor.

With an increase in flight speed, the cone translates rearwards (inwards towards the engine) to maintain the optimal position of the conical (oblique) shockwave; the shock cone angle decreases with increasing flight speed. Hence the need for the shock cone to translate axially. At increasing flight speed, internal compression occurs in the engine intake due to the required narrowing of the intake area to enhance pressure recovery.

Herman Oswatitsch invented the centre body cone, which was first used on the MiG 21 fighter and the Lockheed SR-71 Blackbird, for example.

Fore-body inlet, or Oswatitsch inlet are alternate names.

**Centreline** The centreline is a line equidistant from a wing section's upper and lower surfaces joining the leading and trailing edges. Alternate definitions are centreline camber and the centreline of the aircraft's fuselage.

See **Camber and camber line; Diagram 4, Airfoil Terminology.**

**Centreline thrust** The term centreline thrust is given to aircraft with two piston-engines mounted in tandem on the aircraft's centreline. The loss of one engine alleviates the asymmetric thrust condition associated with twin-engine aircraft with wing-mounted engines. The term does not apply to single-engine aircraft or jet fighter aircraft with fuselage-mounted engines.

The Cessna 336/337 Skymaster is one example of a centreline thrust aircraft. It is also known as the push-pull configuration.

**Centre of dynamic lift** An alternate name for the aerodynamic centre. The location on the airfoil's centreline where the lift force acts due to forward motion.

*See Aerodynamic centre (a. c.).*

**Centre of gravity (c.g.)** The centre of gravity is the theoretical balance point on a body (aircraft) where the total mass is assumed to act. The centre of gravity is measured from the given datum.

An aircraft has different weights located on different parts of the fuselage and wings. For example, the weight of the fuel, baggage, cargo and passengers. All of these different weights can be assumed to act through one point only – the centre of gravity. It is in fact the point of balance, known as the pivotal point or fulcrum. On an airplane, it is the pivotal point of the aircraft's three axes – the lateral, longitudinal, and normal axes about which, the aircraft rotates. The centre of gravity is very important in the design and operation of any aircraft. Just to complicate matters, the centre of gravity does not remain stationary in any one position but moves around with different loads on the aircraft. Maintaining the centre of gravity within its prescribed fore and aft limits is extremely important in order to maintain the airplane's positive and static stability. The location of the lateral centre of gravity can also play an important part in some aircraft loading situations, usually helicopters and large aircraft.

A forward centre of gravity is more stable in pitch as opposed to a less stable aft centre of gravity location, where the restoring moment is reduced due to less distance between the centre of gravity and the tailplane. The centre of gravity location does not normally affect the lateral stability.

Elevator control authority, static stability and trim all play an important part in determining the centre of gravity range for safe aircraft operation, which in turn determines the degree of stability and 'stick force per 'g'.

The centre of gravity and centre of pressure (lift) turning couple determines the pitching moments and hence static stability. The degree of stability required depends upon the purpose for which the aircraft was designed, for example, fighter, transport or light aircraft, etc.

The centre of gravity for canard or multi-surface aircraft is located at the centroid or balance point between the main wing and the canard wing.

*See Diagram 12, Centre of Gravity Range; Manoeuvre point.*

**Centre of gravity limits** The aircraft manufacturer sets the fore and aft centre of gravity limits to ensure safe flight operations. They are expressed as a certain distance aft of a given datum, as a percentage of the mean aerodynamic chord (MAC) or measured in inches or millimeters aft of the datum to the fuselage station, which is the more familiar term used by pilots of light aircraft. The term percentage mean aerodynamic chord (% MAC) is a useful term to define the centre of gravity limits for larger aircraft. The datum is used as the origin of reference basis from which all fuselage stations are measured and maybe taken from any convenient location on the aircraft. This could be the aircraft's nose, or more commonly, the forward face of the firewall or even some arbitrary point ahead of the airplane's nose.

*See Diagram 12, Centre of Gravity Range.*



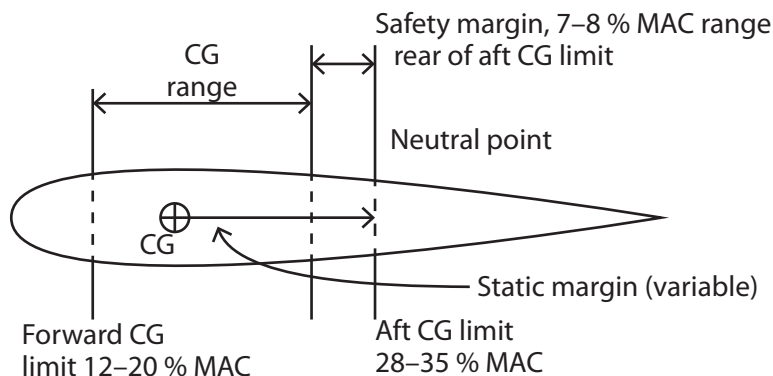
**Centre of gravity safety margin** The centre of gravity margin is the distance from the CG to the neutral point expressed as percentage of the mean aerodynamic chord (MAC) and measured along the longitudinal axis.

See *Diagram 12, Centre of Gravity Range*.

**Centre of gravity range** The CG range for an average light aircraft, is approximately 12–20% MAC for the forward limit and 28–35% MAC for the aft limit. The aircraft must be loaded between the fore and aft centre of gravity limits at all time to ensure the safe operation and ease of handling. Vertical centre of gravity limits may also apply to some large helicopters when carrying out sling load operations.

The centre of gravity range is much larger with power off, but that is only of academic interest and not of much help in loading and operating the aircraft. However, the range can also be reduced to allow for changes in the lift's centre of pressure and the location of the landing gear or flaps when extended or retracted, etc. Longitudinal stability and control force limits determine the centre of gravity range.

*Diagram 12, Centre of Gravity Range* shows the forward and aft centre of gravity limits, the safety margin, and the neutral point in relation to the centre of gravity limits.



**Diagram 12, Centre of Gravity Range**

**Centre of gravity travel** The fore and aft movement of the centre of gravity in flight due to changing fuel loads, etc, is known as the centre of gravity travel.

**Centre of lift** An alternate name for the wing's resultant centre of pressure.

**Centre of pressure** The centre of pressure of lift is considered to be a point on an airfoil's chord line, located at approximately 25% MAC from the leading edge, through which the resultant force (total reaction) acts, divided into the vectors of lift and drag. The centre of pressure may also be considered to be a line from wingtip to wingtip across the span. The location of the centre of pressure and the centre of pressure movement is governed by the airfoil's angle of attack and shape, air viscosity, and compressibility. It is akin to the aircraft's centre of gravity.

The centre of pressure moves forward up to 20% MAC with increasing angle of attack up to the critical angle; the centre of pressure then moves quickly rearwards. First, consider the wing alone theoretically removed from the aircraft. When the wing is in balance the lift and weight forces will be aligned and no turning couple will exist. An increase in angle of attack below the critical angle produces an increase of lift over the wing, especially over the wing's forward section. This in turn causes the centre of pressure to move forwards towards the leading edge, reaching its most forward position around 20% MAC. As the centre of pressure moves forward, the lift and weight forces will be out of alignment and will produce a turning couple resulting in a de-stabilizing pitch-up movement with increased angle of attack. The result is the wing turns end-over-end and will not enter a stabilized glide. Of course, a de-stabilizing pitch-down result will occur if the centre of pressure moves aft. The movement of the centre of pressure is therefore inherently unstable. *Diagram 13, Centre of Pressure Movement*, shows the location and movement of the centre of pressure with varying angles of attack.

If we now replace our fictitious wing back on the airplane, the centre of gravity must now be located forward of the centre of pressure at all times to oppose the wing's unstable pitch-up. This un-stable pitch up must be counteracted by the tailplane in conjunction with the four forces of thrust and drag plus lift and weight all being in equilibrium. The forces of thrust and drag produce a couple which tends to pitch the nose upwards (power is destabilizing) and the lift/weight couple tends to pitch the nose downwards. When these two opposing couple are in balance, the wing pitching moments are zero and therefore stabilized with the centre of gravity location located forward of the centre of pressure.

When a cambered airfoil experiences an increase in angle of attack and hence lift coefficient, the centre of pressure moves forward and stops just before the stall, followed by a rearward movement after the stall. However, the centre of pressure of a symmetrical airfoil does not move (or has very little movement) as the angle of attack changes. All movement is considered to act along the chord line.

The centre of pressure moves due to an increase in angle of attack. Greater upwash in front of the wing creates greater camber over the forward section of the wing's upper surface, which increases the suction and hence, more lift. In addition, due to the angle of attack increasing, the adverse pressure gradient covers more of the wing area, which spreads forwards with turbulent flow and flow separation. The boundary layer finds it more difficult to stay attached to the wing and cannot follow the wing's curvature so easily due to a loss in the flow's kinetic energy and thus, a loss of lift occurs on the aft area of the wing due to the separated flow. Because the separation point moves forward, only the forward part of the wing is producing lift and the centre of pressure moves forward. Therefore, the centre of pressure movement is related to the separation point movement.

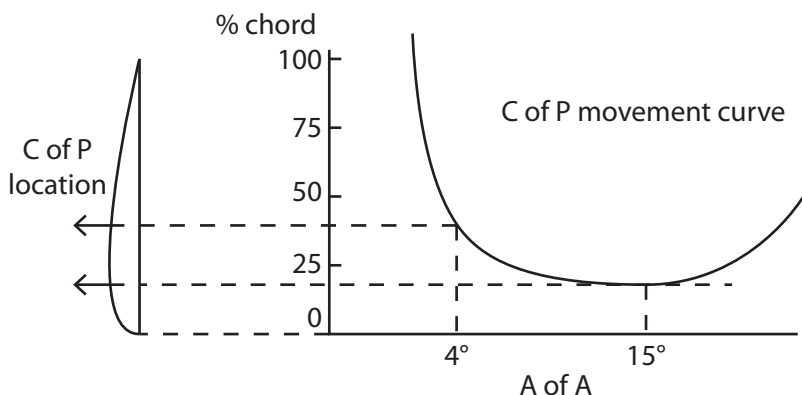
Now, consider the centre of pressure on a three-dimensional sweptback wing during a stall condition. A sweptback wing commonly stalls at the wingtips first with the inner portion

remaining un-stalled. Because of the loss of lift at the wingtips, the centre of pressure moves forward towards the leading edge and inboard laterally on both wings. The mean centre of pressure (on the fuselage centreline) moves forward and hence, closer to the centre of gravity reducing the effect of the nose-down moment and allowing the nose to pitch up further into a deep stall. This is the reason for stick-shakers and stick pushers on jet transport aircraft, to prevent any further pitch-up.

The centre of pressure and the aerodynamic centre should not be confused. The aerodynamic centre is located at a fixed position within the airfoil section profile while the centre of pressure is assumed to be located on the upper surface. The centre of pressure on a cambered airfoil is located further aft than the aerodynamic centre.

The centre of pressure theory was introduced prior to WWI by the German aerodynamicist, Professor Ludwig Prandtl (1880–1953) at the University of Gottingen. Wind tunnel testing during 1915–25 helped Prandtl to understand the centre of pressure's movement over cambered airfoils. Having said all this, the movement of the centre of pressure is not suitable for research purposes, it eventually dropped out of favor to be replaced instead, by the use of the aerodynamic centre located at the 25% MAC position and the associated lift, drag and moments. However, the centre of pressure is still of great interest to all pilots. Professor Samuel S. Langley (1834–1906) discovered the movement of the of pressure on a flat plate moved rearwards with increasing angle of attack and that it must be located forward of the centre of gravity on an aircraft. The Wright brothers carried out airfoil research in their own wind tunnel during 1901–2 when they recorded the centre of pressure moved forward with increasing angle of attack, the opposite way to a flat plate.

**Diagram 13, Centre of Pressure Movement**, shows a graph of the centre of pressure's location and movement with changing angle of attack on an airfoil section.



**Diagram 13, Centre of Pressure Movement**

**Centrifugal Force** The force acting opposite to centripetal force is the centrifugal force, the word is taken from the Latin verb 'to flee'.

The centrifugal force (angular acceleration) is a rotational inertial force acting away from the centre of the turn. When traveling in a car for example, the car follows a sharp curve in the road but your body resists the turn and tries to continue in the original direction. The force that your body feels is the centrifugal force. It is of course, less obvious when flying in an aircraft due to the banked attitude as it turns counteracting the sensation of the centrifugal force. However, the centrifugal force is still there.

It should also be remembered, when the aircraft is turning, it is not in equilibrium. This is due to it accelerating towards the centre of the turn with air speed constant, even though the centrifugal force is equal and opposite to the centripetal force. Newton's second Law of Motion is related to the centripetal force while Newton's third Law of Motion is related to the centrifugal force, in agreement with the statement 'for every force there is an equal and opposite force'.

*See Turning; Diagram 91, Turning Forces.*

**Centrifugal turning moment** A component of the centrifugal force acting on the propeller blades causes a turning moment about the pitch change axis, which tends to turn the propeller blades towards fine pitch.

**Centripetal acceleration** The centripetal acceleration is the force required to turn an aircraft in flight and is determined by the amount of velocity squared and the turn radius ( $v^2/r$ ).

$$\text{Centripetal acceleration} = v^2/r$$

Where  $v^2$  = velocity squared in meters per second, or feet per second  
 $r$  = radius in meters, or feet.

**Centripetal Force** The centripetal force is the inward force towards the centre of a turn and is equal and opposite of the outward acting centrifugal force.

An aircraft is turned in flight due to the acceleration towards the centre of the turn. The acceleration is produced by the horizontal component (centripetal force) of the aircraft's lift force when the aircraft is banked into the turn. [The name centripetal is taken from the Latin verb 'peter' meaning 'to seek']. The centripetal force acts at right angle to the direction of flight and has no affect on the forward motion of the aircraft. Assuming the aircraft's speed remains constant, the only acceleration is that towards the centre of the turn ( $mv^2/r$ ). It should be appreciated that the centripetal force is a required or inferred force, not an actual force in its own right; it is a vector of the wing lift force. Centripetal forces are kinematic forces as opposed to their counterpart the centrifugal force, which is a kinetic force.

The centripetal force is proportional to the velocity and mass of the moving body, and inversely proportional to the turn radius. This is shown by the formula below. Newton's second Law of Motion can be applied to an aircraft in a turn, which states briefly 'a rate of change in momentum will be in the direction and in

proportion to the magnitude of the applied force'. This translates simply to mean, 'the greater the required acceleration, the greater must be the applied force'. If an aircraft is required to turn at a greater rate of turn, a greater angle of bank will be required along with increased lift to produce a greater horizontal component of the lift vector. Given the aircraft's weight, velocity, and radius of turn, the centripetal force towards the centre of the turn can be found by using the following formula with compatible units:

$$\text{Centripetal Force (CPF)} = mv^2/r \text{ Newtons or } wv^2/gr$$

Where CPF = Newtons or pounds

m = mass in kg (or pounds)

w = weight in kg (or pounds)

$v^2$  = velocity in  $m/s^2$  (or  $FPS^2$ )

r = radius of turn in meters (or feet)

g =  $9.81 m/s^2$  (or  $32.2 FPS^2$ ).

See **Turning**; *Diagram 91, Turning Forces*.

**Centroid of pressure area** The centroid of pressure area has a similar meaning to the wing's centre of pressure. However, the centroid of pressure area applies to a control surface, which is the location of the centre of pressure of the combined upper and lower surfaces acting on the control surface.

**Centroid of surface area** This is the geometric centre of a control surface's area.

**Characteristics of airfoils** The aerodynamic characteristics of an airfoil are dependant mainly on the following factors:

- taper ratio
- aspect ratio
- sweepback angle
- L/D ratio, coefficients of lift, drag and moment
- location and movement of the centre of pressure
- effect on the aircraft's stability
- effect on the aircraft's control harmony and response
- effect on the aircraft's manoeuvrability
- the ability to withstand manoeuvre induced stress.

**Charles' and Gay-Lussac's Law** The relationship between temperature and volume at a constant number of molecules and pressure is called Charles' and Gay-Lussac's Law. At a constant pressure, a volume of gas varies in direct proportion to its absolute temperature. Charles also stated a perfect gas would increase its pressure by  $1/273$  for every degree Celsius rise in temperature provided the volume remains constant. Conversely, for a gas at constant pressure, its mass is directly proportional to the absolute temperature. The equation, known as the 'Pressure law' is given here as:

$$P/T = K \text{ (constant), or written as } P_1/T_1 = P_2/T_2.$$

Jacques Alexander Charles (1746–1823) a French physicist further developed Boyle's findings circa 1787 and Joseph-Louis Gay-Lussac (1778–1850) a French chemist and physicist, verified it in 1802 with more accurate results. Charles also discovered the absolute zero temperature achievable was minus  $273.16^{\circ}\text{C}$ .

See **General Gas Law; Boyle's Law; Gay-Lussac's Law; Ideal gas law; Avogadro's law.**



**Chine** A chine is a highly swept back, low aspect ratio, sharp leading edge wing root extension alongside the aircraft's fuselage and merged with the wing.



**Photo:** The McDonnell Douglas F-18A Hornet has a very prominent chine stretching from the wing root forward to the cockpit area.

The purpose of the chine is twofold:

- First, at high Mach numbers, the sweptback or delta wing's centre of pressure moves aft causing a nose-down pitching moment. The chine produces lift and hence, nose-up trim at the front of the aircraft to counteract the wing's nose-down trim.
- Second, sweptback or delta wings do not perform too well during low-speed, high angle of attack flight. The chine produces a large and strong vortex over the wing root area, which decreases the stall speed and enhances the lift and manoeuvrability of jet fighter aircraft. For example, they adorn the Northrop F-5 Freedom Fighter (the first aircraft to use chines), the McDonnell

Douglas F-18A Hornet, the Russian Sukhoi SU27 Flanker jet fighter aircraft, and other types. Concorde also has a chine that is beautifully molded into the wing's leading edge.

See **Vortex lift; Forebody strake; Strakes.**

**Chord** The wing's chord is the distance from the leading edge to the trailing edge. Apart from constant chord wings, the chord normally reduces from the wing root to the wing tip, although it is not unknown, but rare, for it to increase towards the tip. Measurements are calculated as a percentage of the chord.

The term chord should be further defined as the root, tip, aerodynamic, average, or mean aerodynamic chord.

*See other references to chord.*

**Chord line** The chord line is a reference line joining the leading edge (at the point of minimum radius) and the trailing edge of a two-dimensional airfoil section. The angle between the chord line and the relative airflow defines the angle of attack. It can be considered as the direction in which the aircraft 'points' in flight.

With a symmetrical airfoil, the chord line passes through the centre of the wing section. It may also be taken to be a tangent to the lower surface on airfoils with no cambered lower surface. The chord and camber lines only coincide on symmetrical airfoils.

*See Diagram 4, Airfoil Terminology.*

**Chord length ratio** The chord length ratio is the ratio of the chord length at the maximum width of all propeller blades combined, and divided by the propeller's diameter.

**Cierva, Juan de la** Juan de la Cierva (1895 – 1936) was born in Murcia, Spain. He graduated as a civil engineer in 1919 and became interested in aviation at an early age. By 1913, he had built several model aircraft and later, two gliders and two powered aircraft before becoming the inventor of the Autogiro, which first flew in 1923. After moving to England, Cierva was employed by A. V. Roe (the aircraft manufacturer) in 1925. He was tragically killed in a Douglas DC-2 crash at Croydon, England.

Cierva initially developed the Autogiro as an alternate idea to the helicopter. His first successful machine was the Cierva C4 Autogiro, which on 9 January 1923 made the first officially recorded successful flight of an Autogiro in Madrid, Spain. From Cierva's inventions and his fully articulated rotor blades, the advancement of the helicopter as we know it today became possible.

The modern day autogyro is now known as a Gyrocopter.

*See Autogyro.*

**Circulation theory of lift ( $\Gamma$ )** Aerodynamicists find calculating the circulation to find the downwash and hence the lift, is easier than calculating the distribution of the wing's surface pressure. The convenience of using the circulation theory of lift was a major aerodynamic discovery and it became the forerunner of the lifting line theory.

The circulation of lift does not mean the airflow rotates under and over the wing in a circle; rather it refers to the difference in boundary layer airflow velocity – increasing over the upper surface and a relative decrease along the under surface. The aerodynamic lift is equal to the product of the circulation, wingspan, and velocity and air density in an inviscid flow. The circulation varies in proportion to the angle of attack and air speed; it follows at low speed and high angle of attack the circulation increases and vice versa. It is associated with the wing's bound vortex system. Circulation and vorticity in fluid dynamics are related, but they are two different phenomena.

F. W. Lanchester, Joukowski, and Wilhelm Kutta presented the circulation theory of lift to the world, all independently but around the same time (c1906). Circulation itself is the amount of rotational momentum or energy, added to or superimposed on a passing fluid flow. In accordance with Kelvin's circulation theorem and Helmholtz' vortex theorems, circulation is conserved through the conservation of angular momentum, which requires equal and opposite vorticity. The circulation of the air velocity around an airfoil boundary is calculated by using the mathematical complex line integral method from calculus. The Greek symbol Gamma ( $\Gamma$ ) signifies circulation.

See **Aerodynamicists; Bound vortex; Kutta condition.**



**Photo:** The McDonnell Douglas C-17 Globemaster III in clean configuration flight.

**Classical aerodynamics** Classical aerodynamics is an alternate name for subsonic aerodynamics. It can be divided into the low-speed and high-speed range with 250 KTAS/Mach 0.3 as the accepted border between the two ranges. Above this speed, the effects of compressibility must be taken into account.

Classical aerodynamics was originally known as fluid dynamics.

*See Aerodynamics; Fluid dynamics.*

**Clean configuration** The airplane is in a 'clean configuration' when the flaps/leading edge devices and undercarriage are all retracted.

*See Configuration.*

**Climb performance** It is interesting to note, the maximum lift/drag ratio or initial range speed coincides with the maximum rate of climb speed ( $V_Y$ ) and the endurance speed coincides with the maximum angle of climb speed  $V_X$ . The speed  $V_X$  is approximately half way between  $V_Y$  and the flaps up stalling speed ( $V_{SI}$ ). The maximum angle of climb speed  $V_X$  is only used for obstacle clearance after take-off before accelerating to  $V_Y$  maximum rate of climb speed. During the cruise/climb, the rate of climb in feet per minute will be approximately half way between the rates of climb achieved at  $V_X$  and  $V_Y$ , but at a higher indicated air speed.

*See Diagram 60, Thrust & THP Required & Available Curves.*

**Climbing turn** The difference in speed and angle of attack of the inner and outer wings during a climbing turn causes the aircraft to over bank.

The outer wing is operating at a greater angle of attack, a higher speed and travels a greater distance than the inner wing. This causes the outer wing to produce more lift than the inner wing, resulting in the aircraft over banking. This requires the pilot to apply aileron control to oppose the rolling tendency, or to 'hold-off bank'.

**Clipped wings** Removing a portion of the main wing's tips produces a wing of shorter span known as a clipped wing.

The reduced span compared to the original size wing can have some advantages, especially at relatively lower altitude, where low altitude manoeuvrability (increased roll rate) is improved due to the reduced rolling damping moment. A disadvantage of clipped wings is the lower aspect ratio, which in turn, increases the induced drag and decreases the rate of climb and high altitude performance.

Some Supermarine Spitfires, from the Mark V onwards, had clipped wings; their performance at low altitude (below 25000 feet) was far superior to their German counterparts, the Focke-Wulf FW 190A.

Some air racing planes also perform better with clipped wings, having a greater roll rate when turning around the racecourse pylons.

*See Aspect ratio.*

**Closed circuit wind tunnels** A closed circuit wind tunnel has a return flow circuit. At the upwind end of the test area, the motor and fan are located and downstream of the test area inlet vanes are used to straighten out the airflow as it enters the test area. These types of wind tunnels are far more efficient than the open circuit type; all modern wind tunnels are of the closed circuit type.

**Closed jet wind tunnel** A closed jet wind tunnel has no access to the plane or model while the wind tunnel is in operation, as opposed to an open jet wind tunnel where access is available to the plane or model. A closed jet tunnel may, or may not be an open circuit wind tunnel.

**Cluttered** *See Dirty configuration.*

**Coanda effect** The tendency of a jet efflux of air (or water) to follow a concave surface is known as the Coanda effect (pronounced Kwa:nda) after its discoverer Henri-Marie Coanda (1885–1980). The jet efflux of air captures the surrounding ambient air (known as entrainment) and a low-pressure fluid flow attaches to the surface. This is one explanation of how a wing produces lift.



**Photo:** The Boeing YC-14 prototype has its jet engines mounted above the wings to produce a Coanda effect and increase lift. Located at the Pima air & Space Museum at Tucson, Arizona.

This phenomenon is employed, for example, on the McDonnell Douglas 520N NOTAR helicopter to replace the conventional tail rotor. The tail boom is of greater radius than normal and a forced airflow exits the boom via a slit on its side. The airflow follows the contour of the boom and acts in the same way as a tail rotor – it opposes the main rotor torque. It is also used to advantage on some transport aircraft, which employ blown flaps. The Boeing YC-14 prototype transport had its twin jet engines mounted on top of the wing to direct its exhaust over the wing's upper surface to increase the lift through the Coanda effect; the plane was not

put into production. The name Coanda effect was initiated by Professor Theodore von Karman (1881–1963).

The Coanda effect is also known as flow attachment or surface attachment.

*See* **NOTAR**.

**Coanda, H.** The Bucharest, Romania born aerodynamicist, Henri-Marie Coanda (1885–1980) is credited with discovering the phenomenon named after him in 1930 as the Coanda effect.

**Coarse pitch** A high propeller blade angle is referred to as course pitch, which produces a lower engine RPM.

Course pitch is selected to increase the propeller blade pitch angle to maintain the most efficient angle of attack as the airplane accelerates to cruise speed.

*See* **Propeller forces in cruise flight**.

**Coefficients** The coefficients of lift, drag, and moments are dimensionless numbers commonly found in aerodynamic formulas. The airfoil's shape and angle of attack determine the value of the lift and drag coefficient.

In these formulas, the coefficients are multiplied by air density ( $\frac{1}{2} \rho$ ), velocity ( $V^2$ ) and the area ( $S$ ) to find the required force of lift, drag or moments. For example, the coefficient of drag is found by dividing the total drag by the free-stream dynamic pressure and surface area ( $\frac{1}{2} \rho V^2 S$ ). The lift or moment coefficients are found in a similar manner.

*See* **Lift coefficient; Drag coefficient; Moment coefficient**.



**Coffin corner** The coffin corner is the reduced indicated air speed range of an airplane's high altitude performance. The speed range reduces with increasing altitude between the low-speed aerodynamic stall and the high-speed shock stall. A maximum certified altitude limit is set to avoid flying too close to the reduced speed range. This is shown on *Diagram 10, Buffet Boundary*, as the area above the line 'C'.

See *Diagram 10, Buffet Boundary* and accompanying text; **Inertia stall**.

**Coke bottle** See **Area rule**.

**Coleman instability** See **Coleman theory**.

**Coleman theory** A problem associated with helicopters on landing is ground resonance. When the wheels or skids make contact with the ground a violent oscillation of the helicopter may commence, which if left uncorrected can quickly lead to destruction of the helicopter. The problem was recognized by NACA's R. P. Coleman (hence the term Coleman theory) and Arnold M. Feingold. This is also known as Coleman instability.

See **Ground resonance**.

**Collective pitch** A helicopter's main rotor blades can all increase or decrease their pitch in unison and equally (collectively) to alter the rotor disc lift. Adjusting the collective pitch on the tail rotor allows the helicopter to turn (yaw) around the OZ vertical axis. Collective pitch is controlled by the collective, which is raised or lowered by the pilot to increase or decrease the rotor blade's pitch respectively.

**Combat ceiling** Fighter pilots refer to the combat ceiling as the altitude where the aircraft's rate of climb has reduced to 500 feet per minute.

*Compare with* **Absolute aerodynamic ceiling**; **Service ceiling**.

**Combined Gas Law** *See* **General Gas Law**.

**Composition of the atmosphere** *See* **Structure & composition of the atmosphere**.

**Compound delta** *See* **Double-delta**.



**Photo:** The Westland Lysander has a prominent compound tapered wing. Displayed at the National Air & Space Museum Udvar-Hazy Centre, Washington.

**Compound tapered wing** The unusual shaped compound wing has reversed taper from the wing root (the chord increases) for a short distance outboard, and then tapers in towards the wing tips. The WWII Westland Lysander observation plane must be one of the most famous planes with a compound taper wing.

### **Compressible computational fluid dynamics**

Compressible computational fluid dynamics (CCFD) is a tool used by aerodynamicists to estimate the formation of wave drag over the wing, and other various aerodynamic calculations.

*See* **Wave drag; Computational fluid dynamics.**

**Compressibility** Below an air speed of 200 KTAS/Mach 0.3, the airflow over the aircraft is assumed a low-speed incompressible flow. The airflow acts like flowing water; the air density effects are negligible as it flows over the aircraft and can be ignored. However, at speeds above 200 Knots, the idea of incompressible flow must be disregarded totally and the effects of compressibility (increasing air pressure and the variation in the relative air density with increasing speed) must be taken into account, which also involves some thermodynamic principles.

When compression takes place, the air no longer acts like flowing water. The change is a gradual process from incompressible to compressible flow as the aircraft's velocity increases. Compressibility causes a change in air density (or volume) due the presence of increasing air pressure. Because no heat is added to the flow and air friction is ignored, the flow is termed an isentropic compressible flow.

During the 1930–40s, the speed of the new breed of fighter aircraft (Hurricane, Spitfire and Me 109, etc,) were quite capable of reaching speeds of 400 MPH in a dive. The then unknown effects of compressibility started to become apparent when several WWII fighter aircraft were lost due to compressibility problems in high-speed dives, particularly during combat flying.

The first aircraft believed to have been lost due to compressibility problems occurred on 17 July 1937, when Dr. Kurt Jodlbauer the pilot of a Me 109 was killed when his plane dived into a lake, in Germany. The Hawker aircraft company flight-tested one of their Hawker Typhoon fighters to research the cause of compressibility. Considerable research was performed in the 1940s by the Royal Aircraft Establishment at Farnborough in England using a Spitfire PR.XI, with high-speed dives to Mach 0.92 being performed. NACA in the USA also carried out research into the cause and effect of compressibility.

The term compressibility was introduced during the 1930s.

*See Diagram 14, Compressibility Correction; Lift coefficient & compressibility.*



**Photo:** A Messerschmitt Me 109 fighter is believed to be the first aircraft to be affected by compressibility, similar to this BF 109G example located at the National Museum of the US Air Force in Dayton, Ohio.

**Compressibility barrier** The term sound barrier (the sound barrier does not exist) was presented to the world in 1935, by the British press after they misunderstood a remark by W. F. Hilton, an aerodynamicist at the National Physical Laboratory in London, England.

Sound in itself is not a barrier, but the drag due to compressibility is! The problem lies in the high wave drag rise in the transonic region experienced by aircraft flying faster than approximately Mach 0.8. A term more appropriate than the sound barrier is the compressibility barrier, which better describes the cause of the problem – not sound but the increase in aerodynamic drag.

Therefore, after more than seventy years of the term sound barrier being around, it is now an archaic term and should be put to rest for good! May the compressibility barrier take its place forever!

*See* **Detached shockwave; Sound barrier; Speed of sound (a).**

**Compressibility buffeting** A high frequency vibrating buffet.

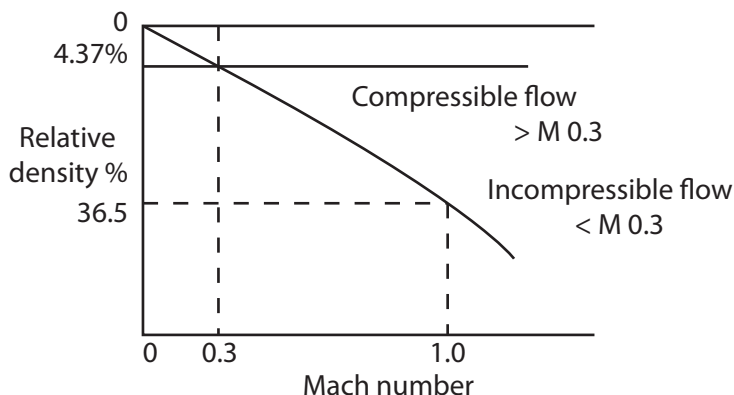
*See* **Buffet.**

**Compressibility burble** The compressibility burble occurs when the airflow breaks away from the wing's upper surface with increasing angle of attack as the air speed reduces.

NACA's John Stack introduced this term in 1933.

**Compressibility correction** In today's modern research and development work in aerodynamics, allowance must be made for high-speed compressible flow on the faster aircraft now flying. Therefore, compressibility correction must be taken into account as the aircraft's speed increases above 200 KTAS/Mach 0.3. This number is used as a 'rule of thumb' by aerodynamicists for correcting low-speed data to high subsonic speed data. Below 200

KTAS/Mach 0.3, the flow is considered incompressible for general purposes.



**Diagram 14, Compressibility Correction**

Pilots normally assume a speed of approximately 200 KTAS/Mach 0.3 (as mentioned above) and an altitude of 10000 feet to be of significance when correcting the air speed indicator reading for compressibility error; (the air speed indicator over-reads with increasing true air speed and must be corrected to show equivalent air speed for navigation purposes. Ambient air density decreases with altitude, but this decrease is over-ridden by the increase in the free-stream air pressure due to the aircraft's increasing speed.

The air becomes more compressible with increasing air speed and altitude and so causes increasing compression at the leading edge stagnation point. The compressibility is less over other parts of the aircraft but it is at the stagnation point and the Pitot tube that is of significance, where the increase in air pressure due to ram effect, increases the relative density. **Diagram 14, Compressibility Correction**, shows the increase in relative air density with speed increasing.

From zero air speed to 200 KTAS/Mach 0.3, the relative air density increases by 5% (4.37% to be exact) with respect to the

pressure change, due to ram effect compression. This is an insignificant amount for most purposes and the airflow can be taken as incompressible. Above 200 KTAS/Mach 0.3, the relative air density error increases considerably until at Mach 1.0, the relative density has increased to about 36.5% above the ambient value. Therefore, the flow is considered compressible above 200 KTAS/Mach 0.3.

The compressibility correction factor was first introduced circa 1922 and published by Herman Glauert (1892–1934) in 1928 as the Prandtl-Glauert correction factor. In 1939, Theodore von Karman (1881–1963) and colleague Hsue-shen Tsein (1911–2009) improved the method known as the Karman-Tsein rule. This method was superseded (post 1939) by the present formula devised by the American Edmund V. Laitone (1915–2000).

*See **Air speed** for corrections from rectified air speed (RAS) to equivalent air speed (EAS).*

**Compressibility effects**      The effects of increasing compressibility are associated with high-speed flight, as listed below:

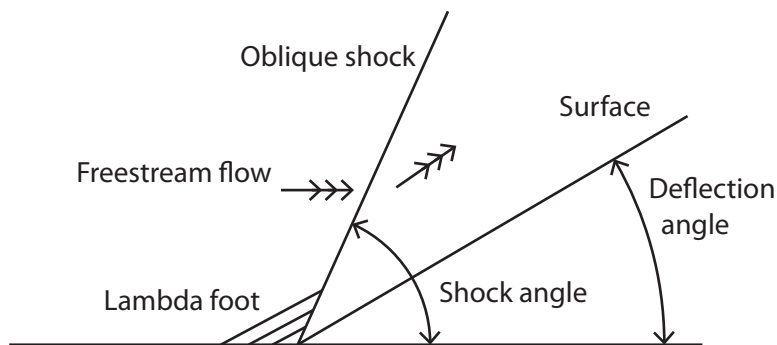
- rapid drag rise, which prevents or reduces acceleration through Mach 1.0
- boundary layer separation causing a loss of lift
- Centre of pressure moves rearwards on the wing causing Mach tuck
- shockwaves begin to appear adding to the drag
- compression lift in supersonic flight can be generated by accelerating the airflow below the wing
- Laminar flow is inhibited causing a decrease in the airflow velocity.

**Compression corner** The term compression corner refers to a convex corner where a supersonic flow is turned into itself inducing a shock wave with a Lambda foot. A compression corner can be found for example, where the air flows over the nose of an aircraft and meets the cockpit canopy and is forced to change direction. It is also known as a wedge.

See **Lambda shockwave**.

**Compression flow** When a supersonic airflow passing over the aircraft meets a wedge-shaped object such as the cockpit canopy, it is forced to turn a corner, and a positive compression flow will form. This causes a reduction in the airflow velocity and a change in airflow direction parallel to the new surface. Although the airflow velocity is reduced, it remains supersonic with the air pressure, density and temperature all increasing due to the air being compressed as it passes through the shockwave. The increase in temperature raises the speed of sound (an increase in the true air speed of Mach 1.0) and the Mach angle become more acute.

Most aerodynamic shock waves on the external fuselage are weak shocks rather than strong shockwaves found on the wings.



**Diagram 15, Compression Flow & Oblique Shockwave**



The weak shockwave limits the turning angle, which is less than the turning angle of the airflow that is found in a compression flow. The base of the oblique shockwave is also known as a compression corner where small Lambda compression waves may form.

Small leading edge radii are found on supersonic aircraft wings to prevent a bow shockwave from forming in the compressed airflow, which increases drag substantially.

*Compare with* **Expansion flow & fan; Lambda shockwave.**

**Compression lift** Compression lift is due to the shock layer increasing the surface pressure acting on the underside of the wings and fuselage of high-speed supersonic or hypersonic aircraft.

The North American Aviation XB-70A Valkyrie experimental aircraft was designed with actuating wingtips that drooped 65



**Photo:** The BAC TSR-2 prototype relied on compression lift to maintain high speed, high altitude flight.

degrees to obtain compression lift, which had the added benefit of reducing the induced drag by around 30%. Two NACA scientists, Alfred Eggers and Clarence Syvertson at the Langley Research Centre introduced the compression lift concept in 1956.

Also known as wave riding.

**Compression pylon** A specially shaped engine pylon for mounting under-wing engines on a swept-wing jet transport aircraft.

A Mr. Patterson junior invented the compression pylon, as an alternate method to the Kutney bump. It is designed to negate the effects of supersonic flow through the channel formed by a jet transport's fuselage, under wing surface, the inboard engine, and engine pylon. The thickness of the pylon increases from the leading edge to the trailing edge particularly on the inboard side of the pylon. The maximum thickness is reached at the trailing edge.

An alternate name for Kutney bumps.

See **Kutney**.

**Compression waves** Compression waves (also known as whiskers) are small, diffused wavelets formed at the base of an oblique shockwave, which increase the flow pressure on passing through a shockwave. They are the opposite of expansion waves, which decrease the pressure.

See *Diagram 15, Compression Flow & Oblique Shockwave*.

**Computational fluid dynamics** Computational fluid dynamics involves the use of complex mathematics.

Aeronautical engineers and aerodynamicists use a super-computer to calculate the complex, computational fluid dynamic differential equations, which affect the airflow over test aircraft. The method was initially designed to solve the problems of airflow around complex shaped aircraft.

Using a simplified version of the Navier-Stokes equations, the aerodynamicists can create a mathematical model of the aircraft under test using the computer to calculate thousands of points to find the pressure distribution over the aircraft for a given time period. The results to the answers are found in a few hours using computational fluid dynamics that would normally take days or longer to find the answers in wind tunnel experiments.

Computational fluid dynamics is commonly used these days during the design of modern aircraft. For example, the Boeing 737-300s with the then new CFM56-3 fanjet engine cowlings were designed within a very short time using the computational fluid



**Photo:** Computational fluid dynamics was used to design the engine cowling to house the CFM56-3 turbofan engine used on the Boeing 737-300 and later models.

dynamics technique. Without this method, the design work would have been a long drawn out process. The Boeing 737-300 first flew on 24 February 1984 with the CFM56-3 engines. Computational fluid dynamics also enabled refinement to the design of the main wings for the Airbus A380 jet transport. The result was a 2% drag reduction due to the altered camber and wing twist from the wing root to the tip; the wing root trailing edge has a very distinctive downturn. The use of computational fluid dynamics began circa 1962.

**Condensation spirals** Under humid atmospheric conditions, the decrease in air pressure behind a propeller can produce visible spiral contrails. These are formed in the same way as contrails are formed from wingtips. The decrease in air pressure is accompanied by a decrease in air temperature in the core of the contrail, which condenses to form visible trails. Helicopter rotor blades can also form condensation spirals.

**Configuration** Configuration refers to the position of the aircraft's landing gear, spoilers and flaps. The aircraft is said to be in a 'dirty' configuration when configured for landing with flaps and gear down. With flaps and gear up, the aircraft is in a 'clean' configuration.

**Coning angle** The coning angle is the angle made between a helicopter's main rotor blade feathering axis and the plane of rotation. A second definition is the angle between the tip path plane and the feathering axis.

The rotor blade's coning angle varies with changes in the blade pitch angle due to changing flight conditions. The angle increases with greater rotor blade thrust (lift force) acting at right angles to the feathering axis, but is opposed by the centrifugal force caused by rotor RPM, which tends to decrease the angle until a state of balance exists and the coning angle remains constant. It

also depends on the helicopter's gross weight and its g-force. Blade pitch angle is akin to a wing's angle of attack.

*See **Diagram 45, Inflow (Transverse) Roll.***

**Conservation of angular momentum** The momentum is the quantity of motion of a moving body; it is the product of mass times velocity (MV) when acting in a straight line. Momentum of a rotating object is known as the conservation of angular momentum, which can be expressed by the following formula:

Angular momentum = rotational inertia times angular velocity.

If external forces are absent, then momentum is conserved. Likewise, if torque forces are absent, then angular momentum is conserved.

*See **Coriolis force.***

**Conservation of Energy** Energy can neither be created nor destroyed, increased or decreased. Energy is always transformed from one source to another of similar quantity, hence the law of conservation of energy. Although it is true to say, not all energy is converted to perform the required task – some energy is lost in the process. For example, when fuel is burned in a piston-engine it is not all converted to mechanical or kinetic energy, some is lost as heat and sound. Kinetic energy plus heat plus sound energy together equals the potential energy of the fuel (chemical energy).

The conservation of energy law can be shown by the following formula:

$$PE + KE = \text{constant}$$

Where PE = potential energy

KE = kinetic energy.

This formula states the energy of position (potential energy) and the energy of motion (kinetic energy) remains constant. Energy

can be transferred either way between potential energy and kinetic energy with the sum total remaining the same.

Sir Benjamin Thomson (Count Rumford) (1753–1814) an English physicist is credited with introducing the Law of the Conservation of Energy in 1797. Its importance relates to the First Law of Thermodynamics.

*See* **Energy – potential & kinetic; Kinetic energy; Bernoulli's theorem.**

**Constant ( $\frac{1}{2}$ )** The constant half ( $\frac{1}{2}$ ) appears in the dynamic pressure equation ( $\frac{1}{2} \rho V^2$ ), which is also part of the lift and drag formulas.

**Constant-speed propeller** A constant-speed propeller is one with a constant-speed unit, which automatically maintains a chosen engine RPM within the propeller governor's range.

The Boeing 247-D was the first modern airliner, in 1934, to use a constant-speed propeller. The improved performance given by the constant-speed props enabled the Boeing 247D to become the first airliner capable of climbing on one engine. Constant-speed propellers are found on every turbo-prop and all but the basic single-engine light airplanes.

**Constant-speed unit** The constant-speed unit (CSU) located at the propeller hub is the device used to select or maintain a constant propeller/engine RPM within its operating range of 1600–2800 RPM, or thereabouts on a piston-engine.

**Contact discontinuity** The location where the outer edge of a jet aircraft's supersonic exhaust flow meets the free-stream airflow is known as the contact discontinuity. Expansion and compression waves interact with each other as they are reflected from the contact discontinuity (outer edge) and shock diamonds are formed and become visible in the supersonic exhaust flow.

*See* **Shock diamonds.**



**Photo:** The constant-speed unit in the propeller hub of a Boeing B-17G Flying Fortress, located at the Pima Air & Space Museum in Tucson, AR.

**Continuity equation** The continuity equation deals with the steady high-speed fluid-flow where its cross-sectional area, density, and velocity remain constant. In a relatively low-speed flow (below 200 KTAS/Mach 0.3) the effects of density is ignored. When a fluid flows through a narrow constriction, the velocity must increase to maintain a constant flow rate resulting in a decrease in pressure. The airflow velocity doubles if the constriction area is halved. Bernoulli's theorem describes the continuity equation in relation to pressure changes. This is shown by the following formula:

$$\text{Constant} = AV\rho$$

Where A = inlet area

V = airflow velocity

$\rho$  = air density.

Below 200 KTAS/Mach 0.3, the formula becomes:

$$\text{Constant} = AV.$$

It is also known as the law of continuity or conservation of mass and is related to incompressible flow.

See **Bernoulli's theorem**.

**Continuum flow** A continuum flow involves the action of random molecular motion, as found in dense air. Due to their close proximity, the flow consists of a constant bombardment of the molecules and is therefore called a continuum flow. Most aircraft flight and the study of aerodynamics occur in a continuum flow. The opposite of a continuum flow is a free-molecular flow, which is found in hypersonic flows.

Compare with **Free-molecular flow**.



**Contra-props** A contra-rotating propeller is one that consists of two co-axial mounted propellers and driven by the same engine, but rotating in opposite directions. By using two propellers mounted on the same shaft with a given propeller diameter, the total amount of power absorbed can be greatly increased. The rear-mounted propeller in the pair straightens out the helical slipstream of the first propeller, thereby reducing the propeller torque to almost zero. The lack of torque reaction has the advantage of reducing the yaw on take-off and also in-flight yaw and roll caused by power changes during flight manoeuvres, which is an important factor on high-powered aircraft.

Co-axial prop is an alternate name.



**Photo:** The Avro Shackleton bomber has four engines each powering a set of contra-props. This model is in the Manchester Air Museum, England.

**Control buffet** See Pre-stall buffet.

**Control horn** On a helicopter's main rotor system, the control horn is a method to decrease the blade angle when the blade pitches up, or vice versa. It is an alternate name for offset pitch horns.

*See* **Offset pitch horns.**

**Controllability** The degree of controllability depends on how well the aircraft responds to control inputs by the pilot to adjust the flight path. It is associated with stability where the upper limit of aircraft control determines the amount of stability and vice versa. For example, jet fighter aircraft are more manoeuvrable than a jet transport, which has greater stability

**Control power** Control power is the ability of the cyclic control to maintain the helicopter's longitudinal and lateral stability and to reduce the forces of pendulosity or swinging of the fuselage under the rotor disc. Keeping the fuselage and rotor disc in alignment as much as possible is essential to maintain stability and control.

Offset flapping hinges and various types of rotor systems are employed to promote control power. For example, the rigid rotor has the most control power, the semi-rigid has the least, while the fully articulated system has control power lying mid-way between the fully and semi-rigid rotor systems.

*See* **Pendulosity.**

**Control reversal** Control reversal can occur during high-speed flight, especially in a dive, when the wing deforms during aileron deployment.

*See* **Aileron reversal; Mach tuck.**

**Control surfaces** Moving the control surfaces from their streamlined position to deflect into the airflow imposes a load on the wings or tail to alter the airplane's attitude. Control surfaces can be divided into primary and secondary groups.

The primary controls allow the pilot to control the aircraft's roll, pitch and yaw, around its three axes; these are the ailerons, elevators and rudder respectively. Elevons and ruddervator are also included in this group. The secondary controls are the flaps, spoilers and leading edge devices. These secondary controls modify the wing's geometry to change its low-speed lifting ability, usually for take-off and landing. The exception is the spoilers, which can also be used during high-speed flight to reduce the aircraft's speed, or differentially to aid aileron control.

**Convergent duct** Subsonic turbojet or turbofan-powered aircraft have engine exhaust nozzles in the shape of a convergent duct.

The purpose of the duct is to increase the velocity of the exhaust (and hence thrust) as it does in the common venturi tube. The exhaust velocity must always be higher than the aircraft's forward speed in order to provide forward thrust. Although the aircraft itself may be flying at a very high, subsonic Mach number, the exhaust velocity does not reach the speed of sound because the speed of sound increases with the higher temperature in the exhaust.

See **Convergent/divergent duct; Divergent intake duct; Ram effect/recovery; Venturi.**

**Convergent/divergent duct** The convergent/divergent duct (also known as a con/di nozzle) is a venturi shaped tube designed for use in high-speed airflows, to convert pressure energy into kinetic energy, and vice versa in the divergent section.

They are commonly found on the jet engine tail pipes of transonic or supersonic aircraft due to the higher velocity (transonic) exhaust gas exiting the tail pipe, The first part of the duct is convergent (the streamlines converge) which increases the velocity of the exhaust gas to supersonic speed until it reaches the duct's throat. The pressure energy is then converted into kinetic energy. Downstream from the throat, the duct is divergent (expands) due to the greater

volume and velocity of the exhaust gases ejected from the tailpipe. The streamlines diverge through the opening duct, which decrease the gas flow speed and converts the kinetic energy into pressure energy.

Careful design of the convergent/divergent duct is required to ensure the gas velocity/volume is contained within its limits to avoid any loss in thrust efficiency. Subsonic aircraft have the intake of their gas turbine engines formed in the shape of a divergent duct to reduce the intake velocity and increase the air pressure before entering the compressor. Between the intake and exhaust system, various parts of a gas turbine engine are designed with converging or diverging ducts to control the airflow and maximize the output efficiency of the engine.

A convergent/divergent duct is also used on some high-speed jet aircraft engine intakes to reduce the supersonic inflow velocity to a subsonic velocity before it enters the engine's compressor.

*See* **Convergent duct; Divergent intake duct; Ram effect/recovery; Venturi.**

**Cooling drag** The airflow required to carry away engine heat to atmosphere can be a source of considerable drag. Careful design of the air ducts through the engine bay can reduce the drag to an acceptable limit.

**Cooper-Harper rating scale** NACA flight-test pilot George Cooper devised a numerical system to evaluate the handling qualities of US military aircraft during flight-testing at the Ames Aeronautical Laboratory, at Moffet Field in California. The scale rates aircraft handling qualities from one to ten, with one being the best quality and ten being unacceptable. The rating scale was first published in 1957 as the Cooper Pilot Opinion Rating Scale. The system was improved upon in 1969 by Robert Harper from Cornell Aeronautical Laboratory and became known as the Cooper-Harper rating scale.

*See* **Flying quality.**

**Coriolis force** The Coriolis force acting within a rotating system is an apparent force with a radial velocity. It is referred to as a force but it is in fact a pseudo force, or fictitious inertial force. No force actually exists; it is more of a reaction, or effect, than a true force. However, it is attributed to the rotation of a mass. The angular momentum of the rotating mass remains constant or, in other words, it is conserved. A change in one factor (rotational inertia or angular velocity) results in a change in the angular momentum or Coriolis force. Simply, an increase in the angular velocity of the mass produces a decrease in the radius of rotation and vice versa.

A good example of this is the spinning ballerina. Stretching her arms above her head reduces her body's radius of mass causing her to spin faster caused by her rotational inertia decreasing with constant angular momentum; so therefore, her angular velocity (spin rate or RPM) increases. Stretching her arms out horizontally increases the radius of mass and slows down her angular velocity or rate of spin.

A helicopter's rotor blade has its centre of mass about half way along the blade. If the blade flaps up due to dissymmetry of lift the centre of mass moves inboard towards the blade hinges. Due to the conservation of angular momentum, when the blade flaps up the rate of rotation (RPM) increases.

The Coriolis force on a three-blade rotor system is greater than on a two-blade system with under slung blades, which reduces hunting. By allowing the blades to flap up, vibration and damage is reduced. The Coriolis force is also greater on larger helicopters. The hunting needs to be controlled by absorbing the movement in some way. Flexible blade roots, hydraulic or friction dampers or lead-lag hinges are different methods employed to control hunting due to the Coriolis Effect.

The Coriolis force was named after its discoverer Gaspard de Coriolis (1792–1843) a French physicist in 1835.

See **Conservation of angular momentum.**

**Corner speed/velocity** The corner speed or velocity is the speed at which the aircraft achieves maximum performance in the turn. The g-force is at its maximum limit with the true air speed at the  $V_A$  manoeuvring speed. A higher speed reduces turn performance by increasing the turn radius. Therefore, there is only one corner speed to balance the requirement of maximum lift for minimum radius without overstressing the aircraft. Turning at a speed above or below the corner speed will reduce the aircraft's turning performance.

The point on the V-n manoeuvre/gust envelope where the stall line intersects the line for the limit load factor at the aircraft's  $V_A$  a manoeuvring speed is known as the corner speed.

See **Manoeuvre speed ( $V_A$ ); Diagram 59, Manoeuvre Envelope.**

**Counter-rotating propellers** Twin-engine airplanes have the propellers of each engine rotating in opposite directions with the top blade (normally) turning towards the fuselage. With this arrangement, the propellers are known as counter-rotating propellers.

The main advantage with this configuration is during take-off and climb-out after an engine failure. On a conventional twin-engine aircraft with both propellers turning clock-wise, asymmetric thrust causes the greatest yaw when the left-hand engine is shutdown. This is because greater thrust is generated on the down-going side of the propeller disc; remember the P-factor or asymmetric disc loading? On a right-handed propeller (rotating to the right), this places the centre of thrust to the right of the propeller axis.

On a conventional twin-engine aircraft (without counter-rotating propellers), the centre of thrust on the left-hand engine will be closer to the aircraft's normal axis, but on the right-hand engine, the centre of thrust will be further away from the OZ axis. In other words, the centre of thrust is offset to the right of the propeller's axis on both propellers. In the event of a left-hand engine failure, the right-hand engine will cause the greatest yawing

moment. Not forgetting, the drag of a windmilling propeller will also contribute to the yawing force. In the case we are considering here, the left-hand engine is said to be the critical engine, because of the greater yaw force and handling problems involved. Airplanes with propellers rotating anti-clockwise, or 'left-handed' propellers, will have their right-hand engine as their critical engine. It should be obvious here, that control problems are more critical when the 'critical' engine is shut down.

**Couple** A couple is a pair of parallel forces acting in an equal but opposing direction. Because a couple does not act through a single point, the system tends to cause rotation. A good example being the two pairs of couples found in aerodynamics to describe the vertical positions of lift and weight and the horizontal position of thrust and drag of an airplane. The two pairs of couples help to keep the airplane in balance. If one of the forces in a couple is reduced, say the thrust force, then the resulting rotation caused by the other couple (lift and weight) will cause the airplane's nose to pitch down and vice versa.

The moments of a couple are calculated by multiplying one of the forces in Newtons by the distance between them in meters, the answer being in Newton-meters (N-m) as for torque.

**Coupling** Coupling is the interaction between movements in two planes. Yaw and roll is a good example; if movement occurs in one plane, it induces movement in the other.

It is also known as cross coupling.

**Cranked wing** *See Inverted gull wing.*

**Crescent wing** A type of modified swept-back wing where the angle of sweep varies in distinct stages reducing from the wing root to the wingtip.

On a fully sweptback wing, the airflow can tend to flow spanwise towards the wingtips, stagnate, and result in tip stalling. The purpose of the crescent shaped wing is to reduce the effects of the spanwise flow of air towards the wingtips. The wing's inboard half-span is greatly sweptback while the outboard half-span has noticeable less sweepback. The inboard sweepback is ideal for high-speed subsonic flight. Therefore, the crescent wing benefits from both sweepback and (relatively) straight wings.

The crescent shaped wing dates back to WWII when research work by the German companies Arado and Blohm & Voss culminated in Arado's, Rodiger Kossin's design of the Arado AR-234 jet bomber. The bomber's crescent wing had a  $37^\circ$  sweep at the root and  $25^\circ$  of tip sweep.



**Photo:** The Victor jet bomber seen here at the Imperial War Museum, Duxford, England, was developed by Handley-Page with a crescent shaped wing.



Handley Page built the HP-88 research aircraft (first flight 1951) to flight-test the crescent shaped wing designed for the Handley-Page HP-80 Victor, which was built for the Royal Air Force V-Bomber fleet. The German, Dr. Gustav Victor Lachmann (1896–1966) designed the Handley-Page Victor and gave his name to it. The Victor made its first flight on 24 December 1952.

The crescent shaped wing has a sweep angle and chord that decrease in three distinct stages from root to tip. The low thickness/chord ratio produces a low drag airfoil with long range and an extremely high operating altitude. This design ensures a high critical Mach number is maintained along the entire span.

Crescent wings are also known as scimitar wings.

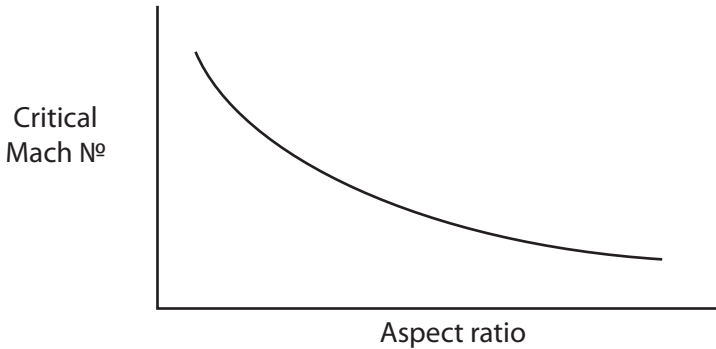
**Critical angle of attack** The angle of attack at which the lift peaks and the airflow becomes turbulent and breaks away from the wing at the separation point followed by the aircraft's nose pitching-down with a partial loss of control.

Also known incorrectly, but commonly, as the stalling angle.

See **Critical angle of attack**; *Diagram 49, Lift Coefficient V. Angle of Attack.*

**Critical engine** In the event of an engine failure on a multi-engine aircraft, the critical engine is the one that produces the most adverse yaw when it fails. Due to the propeller's offset centre of thrust, the critical engine will produce more yaw than the loss of an engine on the opposite side of the aircraft. The loss of the critical engine on take-off determines the  $(V_1)$  go/no go decision speed.

**Critical Mach number** The critical Mach number ( $M_{\text{CRIT}}$ ) refers to the free airstream Mach number at which the drag-rise increases noticeably due to the commencement of transonic flow at around Mach 0.87.



**Diagram 16, Critical Mach Number V. Aspect ratio**

An aircraft flying at high subsonic speeds may attain a speed at which the airflow over the wing reaches Mach 1.0, due to the camber (curvature of the wing); this speed is known as the critical Mach number. It is the thickness/chord ratio and degree of sweepback that determines the critical Mach number of an airfoil and the amount of drag at transonic speed. At the critical Mach number large changes in forces, moments and pressures on the wing takes place, including possible airflow buffet and boundary layer separation if the shock wave is sufficiently strong enough.

The critical Mach number and lift coefficient are closely related; if a high Mach number is desired a wing with a lower maximum lift coefficient will be required. This is achieved on wings with lower thickness/chord ratio to reduce the velocity over the upper surface, which increases the critical Mach number. Supercritical wings designed for transonic flight have the maximum thickness located well aft combined with a small leading edge radius.

At a speed slightly higher than the critical Mach number, a shockwave will form on the wing's upper surface causing an increase of drag, a decrease of lift, the centre of pressure moves aft followed by airframe buffeting and eventually a possible loss of control. If the aircraft's cruise speed is at Mach 0.87 and the airflow over the wing reaches Mach 1.0, then the Critical Mach number (for the aircraft) is Mach 0.87.

In 1918, two engineers, Frank W. Caldwell and Elisha N. Fales at Mount Cook Field (later Wright-Patterson Air Force Base) first introduced the term critical speed, which later became known as the critical Mach number. In 1919–20, they built the first high-speed wind tunnel in the USA for propeller research. They proved the critical Mach number is higher for thin wings compared to thick wings.

*See Drag divergence Mach number ( $M_{DIV}$ ); **Diagram 21, Drag Divergence Curve.***

**Crocco, G, A** General Gaetano Arturo Crocco (1877–1968) was a leading Italian pioneer in aerodynamics and astronautics. He was the first person to suggest using a cyclic pitch control for helicopter rotor blades, in 1906. He was also involved in the design of airships and rockets, amongst other achievements.

**Cross coupling** Cross coupling, or simply coupling, is the influence one axis of stability has on another axis. A good example is when the aircraft rolls followed by a variations in the pitch attitude.

*See Inertia coupling.*

**Cross sectional area** A cross sectional area is a transverse, two-dimensional view through an object, such as a fuselage or wing section. It is also known as the frontal area.

**Crosswind force** The crosswind force acting spanwise along the OY axis is positive when acting towards the right-hand wingtip.

**Crossfield, A** Albert Scott Crossfield (1921–2006) was a US Navy pilot, an aeronautical engineer, aerodynamacist, designer and renowned test pilot.

Scott Crossfield was a founding member and fellow of the Society of Experimental Test Pilots and received numerous awards throughout his distinguished career. He attended the University of Washington and served in the US Navy during WW II. In June 1950, he joined NACA as a research test pilot at the NACA High-Speed Flight Research Station, located at Edwards Air Force Base near Los Angeles, California. He played an important part in the Space Shuttle's development and was involved in other space flight projects. He also worked for North American Aviation where he had a great input in the development of the North American Aviation X-15 rocket plane in which he flew the first free flight in 1959. In total, Scott Crossfield flew over one hundred rocket-powered flights to become the most experienced pilot of rocket-powered aircraft as well as several other research X-planes and prototypes.

Scott Crossfield was awarded the Distinguished Service Medal by NASA in 1993 for his many and varied contributions to aeronautics and aviation for over fifty years. For all of his accomplishments he must be best remembered for his record-breaking flights when he became the first person to fly at more than twice the speed of sound (Mach 2.0) in the Douglas D-558-II Skyrocket on 20 November 1953.

Scott Crossfield died on 19 April 2006 when his single-engine Cessna Centurion crashed due to adverse weather conditions, in Georgia, USA.

**Cruciform tail** As its name suggests, the cruciform tail looks very much like a Christian cross with the horizontal tail mounted about half way up the vertical fin. This arrangement keeps the horizontal tail out of the path of the engine's wake. It also reduces interference drag on the fin.



**Photo:** The Northrop F-89J Scorpion sports a cruciform tail. This Scorpion is on display at the National Museum of the US Air Force in Dayton, Ohio.

**Cruise speed ( $V_{CR}$ )** The maximum cruise speed ( $V_{CR}$ ) is shown on the air speed indicator at the top end of the green arc. This is the maximum speed allowed when flying in light turbulence; above this speed, which is in the yellow arc, speed must be reduced to avoid overstressing the aircraft if any turbulence is present. The maximum cruise speed ( $V_{CR}$ ) maybe the same speed as the normal operating speed ( $V_{NO}$ ) unless the normal operating speed is quoted

as being greater, which then applies. The maximum cruise speed ( $V_{CR}$ ) is also marked on the V-n manoeuvre envelope graph and gust envelope curve.

*See* **Manoeuvre envelope; Gust envelope.**

**Cruise thrust** The thrust during cruise flight produced by a piston-engine aircraft can be calculated from the following:

$$\text{Cruise thrust} = PDA \times (V + V_I) \times q \times V_O$$

Where PDA = propeller disc area, sq. meters  
(or square feet)

$V$  = aircraft velocity, m/s (or FPS)

$V_I$  = inflow velocity, m/s (or FPS)

$V_O$  = outflow velocity, m/s (or FPS)

$q$  = air density,  $\text{kg/m}^3$  (or slugs/cubic feet).

*Compare with* **Static propeller thrust.**

**Cusp** A cusp is a depression on the under surface trailing edge of a supercritical airfoil. Its purpose is to assist the high-pressure airflow under the wing to rejoin the lower pressure airflow from above the wing in a smooth flow at the smallest angle possible at the trailing edge. The speed of the two airflows must remain the same where they rejoin at the trailing edge to help delay any shockwave formation on the upper surface from increasing in intensity.

*See* **Kutta condition; Undercamber; Supercritical airfoil.**

**Cyclic pitch control** The pilot controls the helicopter in flight by use of the cyclic-pitch control stick, which is similar in action to an airplane's control stick and works in the same sense.

The cyclic-pitch change is the action of each helicopter main rotor blade to increase and decrease their rotor blade angles of attack independently of each other. Each individual blade changes pitch in order to maintain a constant angle of attack as they rotate

to maintain constant lift over the entire rotor disc. Each blade's angle of attack varies as they rotate into and out of the slipstream, referred to as the advancing and retreating blades.

Changes can be made to the blade pitch angle by two methods. The use of a blade feathering mechanism (a Delta-three hinge) to change the blade pitch angle is known as cyclic feathering. Blade flapping is the alternate method.

It is also known as feathering.

*See* **Blade flapping**.

**Cyclic feathering** The action of changing the helicopter's rotor blade pitch angles is known as cyclic feathering.

# D

**d'Alembert, J** The Frenchman, Jean Le Rond d'Alembert (1717–1783) in 1747 introduced partial differential equations to mathematics. His contributions to fluid dynamics were equally, if not more important, than Bernoulli's contributions. He molded the foundations of theoretical fluid dynamics (classical aerodynamics) in the 18<sup>th</sup> Century.

**d'Alembert's paradox** d'Alembert's paradox was introduced in 1744, due to his study of airflow over a cylinder, where the flow over the top matches the flow over the bottom; the lift cancels out. Likewise, the airflow pressure at the front of the cylinder matches the airflow pressure at the rear; therefore, the drag cancels out. However, in nature, it is impossible to have zero drag, hence the paradox.

His paradox does not apply to supersonic airflow or airflow over finite wings due to the drag induced by the wing (the induced drag). It only applies to infinite wings, that is, wings placed in a wind tunnel where the wing tips abut the tunnel walls.

*See Inviscid flow.*

**Da Vinci, Leonardo** Leonardo Da Vinci (1452–1519) was an Italian painter and designer with interests in a great variety of subjects. In aeronautics, he is noted for his introduction of the law of continuity, the concept of streamlining, designing a theoretical helicopter and stating that air resistance (drag) is proportional to the area of a body, to name a few.

**Dancing diamonds** *See Shock diamonds.*



**Datum** The datum is the reference base from which all stations (or arms) are measured when calculating the longitudinal centre of gravity. Any convenient location on the aircraft may be used as a datum, as chosen by the aircraft manufacturer. On light aircraft, the 'nose' or the forward face of the firewall is more commonly used. For larger aircraft, percentage MAC is a more convenient method for centre of gravity calculations.

Also known as the reference datum.

**Dead air region** At high angles of attack, the high-pressure air from below the wing may flow around the trailing edge on to the upper surface of the wing. This causes the low-pressure air to burble and breakaway with a loss in lift leading to a stalled condition. It is also known as the wing's third region, recirculation flow or reversed flow.

See **Region of reversed flow; Diagram 1, Adverse and Favourable Pressure Gradients.**

**Decalage** The term decalage has three definitions. The first term relates to biplane or triplane main wings, where the decalage is the different angle at which the wings are attached to the fuselage. The decalage angle is measured between the wing's chord line and the fuselage longitudinal axis. Decalage may be positive on one wing and negative on the other; negative decalage improves longitudinal stability.

Decalage on a monoplane can be aerodynamic or geometric. Aerodynamic decalage is measured relative to the main wing's zero lift line and the tailplane's zero lift line. The geometric decalage is measured relative to the main wing's chord line and the tailplane's chord line. The main wings are placed at a greater angle of incidence by about two degrees more than the tailplane. The purpose of decalage is to aid longitudinal stability.

Dr. Maxwell Munk (1890–1986) in his 1923 research at NACA Langley stated decalage is positive when the lower wing has greater incidence than the upper wing. Decalage is French for offset.

**Deceleron** A deceleron is a type of airbrake where each aileron simultaneously can split up and down to increase the aerodynamic drag to reduce speed. It was developed by Northrop and first used on the Northrop F-89 Scorpion jet fighter. The Northrop B-2 Spirit stealth bomber uses them and the Space Shuttle has a split rudder deceleron to act as an airbrake.

See **Spoilers; Speed brakes; Dive brakes; Diverter; Lift dumpers.**



**Photo:** The Northrop F-89 Scorpion was the first aircraft to be built with decelerons. This example is on display in the National Museum of the US Air Force in Dayton, Ohio.

**Deep stall** A deep stall involves very large angles of attack beyond the critical angle of attack. After the stall break, the angle of attack continues to increase rapidly to a point-of-no-return where recovery is impossible. Early T-tail, rear-engine jet transport aircraft were initially affected by this condition. For this reason, most jet transport aircraft now have stick pushers and/or stick shakers installed.

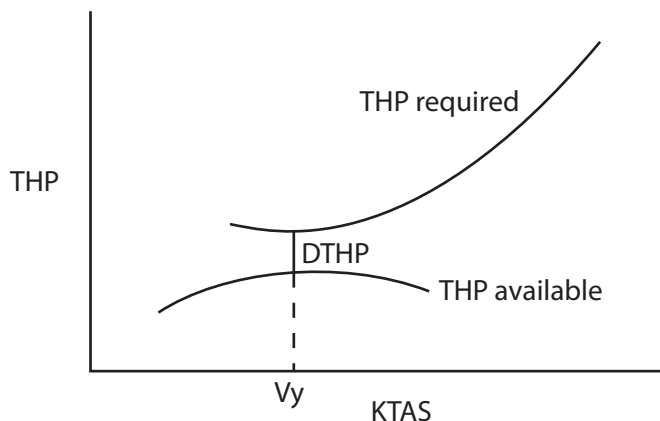
The sweptback wings of a jet transport will stall first at the wingtips. The wingtip vortices then move to a new position further inboard along the span of the wing to the boundary of the stalled and un-stalled area of the wing. The rotation of the vortices strikes the tailplane from above increasing the tail down force, which pushes the nose up further into the stalled condition. Once the pitch-up has started the nose moving upwards, inertia may take over to continue the upward trend into a deep stall.

The Handley Page Victor swept wing jet bomber was the first aircraft type to experience deep stall tendencies in 1962.

Also known as locked-in stall or superstall.

**Deficit thrust horsepower** Deficit thrust horsepower is the power required by the aircraft but is unavailable from the engine. The difference in power required and power available is a negative value known as the deficit thrust horsepower.

With reference to *Diagram 17, Deficit Thrust Horsepower v. KTAS*, when the engine is throttled back the thrust horsepower available is reduced and the curve will descend the graph. When the two curves, thrust horsepower available and thrust horsepower



**Diagram 17, Deficit Thrust Horsepower v. KTAS**

required coincide, the rate of climb will be zero and the aircraft will maintain straight and level flight. Any further power reduction in power the aircraft will descend with the descent rate increasing to its maximum value with the throttle fully closed.

**Degrees of freedom** An aircraft's six degrees of freedom are three rotational (roll, pitch, and yaw) and three translational (up/down, forward/back and left/right).

*See Axes; Diagram 2, Planes of Movement.*

**de Laval nozzle** A de Laval nozzle is a converging/diverging duct (throat). It is commonly found in jet engine exhaust nozzles on aircraft, which operate in the transonic or supersonic speed range; some rocket engines and a few wind tunnels use it.

The de Laval nozzle is similar in shape to the common venturi tube where the entrance to the duct converges to a throat and diverges downstream. If the flow-speed through the nozzle's converging section is in the transonic region, shockwaves may form in the airflow depending on the duct shape causing compression; the velocity will decrease slightly and the pressure, temperature, and density will increase. As the airflow passes through the diverging section of the nozzle, the airflow expands through the expansion waves causing the airflow velocity to increase and the pressure, temperature and density to decrease.

A subsonic, incompressible flow through a venture tube experiences the opposite to the above; the flow-speed initially decreases with pressure increasing on entry. After passing through the throat, the speed increases again and the pressure, decreases back to the ambient value. Density changes very little in the subsonic flow, and temperature change is negligible.

Carl Gustaf de Laval (1845–1913) Swedish inventor, scientist, and steam turbine engineer invented the de Laval nozzle in 1870 when he was developing the steam turbine engine.

Convergence occurs in:

- subsonic flow (venturi) – the flow-speed increases while its pressure decreases below ambient and density remains constant
- supersonic flow (de Laval) – the flow-speed decreases and pressure and density increase with flow compression above ambient conditions.

Divergence occurs in:

- subsonic flow (venturi) – the flow-speed decreases and its pressure rises to ambient with density remaining constant
- Supersonic flow (de Laval) – the flow-speed increases further and pressure, temperature and density decrease with expansion of the flow to ambient conditions.

See **Convergent/divergent duct; Venturi; Compression flow; Expansion flow & fan.**

**Dellane wing.** One of the first tandem wing aircraft designers was Frenchman Maurice Dellane who designed and built the Dellane Duo-mono in 1937. Tandem wings are placed fore and aft of each other along the fuselage, not in the usual biplane configuration.

See **Tandem wing.**

**Delta fins** Delta fins are placed on the rear, lower fuselage of an aircraft for stability purposes. During high angle of attack flight, the delta fins produce some lift causing the tail to rise up and reduce the angle of attack of the main wings to help prevent a stall developing. They also act to damp out any yawing tendencies in turbulence. Delta fins are found on the Gates Learjet 45 XR and the Beech 1900D turboprop commuter, for example.



**Photo:** The Boeing 737 Wedgetail electronic surveillance is one of several types of aircraft with delta fins. Note the three humps at the delta fin/fuselage junction to improve the airflow and reduce interference drag.

**Delta-3 hinge** The Delta-3 hinge is a type of flapping hinge on a helicopter's rotor blade, which attaches it to the rotor hub. Due to its ability to flap, the Delta-3 hinge has the same effect on flapping as do offset pitch horns. It flaps up and down but has no effect on changes in lift coefficient due to the reduced effect of dissymmetry of lift. This is achieved by mounting the hinge line at an angle other than  $90^\circ$  to the feathering axis, as it is on a conventional hinge. The result is, as the blade flaps up its pitch angle is decreased and vice versa. Therefore, flapping hinges are a requirement to provide cyclic feathering, which compensates for dissymmetry of lift.

It is also used on tail rotors due to their location on the tail boom where vibration is a major problem, due to the tail rotor's exposure to varying inflow forces from the main rotor and the natural wind.

The tail rotor being mounted with its axis horizontal, the rotor experiences dissymmetry of lift due to the difference in speed between its advancing and retreating blades.

See **Offset pitch horns; Flapping hinge.**

**Delta wing** The delta wing takes its name from the Greek capital letter delta ( $\Delta$ ), the wing being similar in planform shape to an isosceles triangle, with the base of the triangle representing the wing's trailing edge.



**Photo:** The Fairey Delta 2 was an early research aircraft with a delta wing planform. This aircraft is now located at RAF Museum Cosford, UK.

The low aspect ratio wing has a high sweepback angle of at least 45° or greater, pointed wingtips and a straight trailing edge. The wing has greater area for a given span compared to a straight wing and thus less wing loading and ample fuel storage space. The wing is devoid of flaps, because they would be ineffectual and act like elevators due to their distance from the centre of gravity.

The delta wing has the advantage of relatively good handling characteristics and greater lift at high angles of attack and a lower landing speed than a conventional wing. Aeroelastic problems are of less concern. The delta wing does not stall in the normal sense of the word; it has no stall speed as such. The nearest equivalent to the stall speed is the zero rate of climb speed ( $V_{ZRE}$ ). One disadvantage is the increased induced drag due to the delta's low aspect ratio.

Butlers and Edwards first patented the delta-type wing as early as 1867. However, a successful German aerodynamicist, Professor Alexander Lippisch (1894–1976) pioneered the delta wing and gave us the term delta. The world's first delta wing aircraft, which he designed, flew in 1931 with separate ailerons and elevators for control; elevons came later. Professor Lippisch was the consultant for America's first successful delta wing research aircraft, the Convair XF-92A, the forerunner of the F-102 Delta Dagger that first flew in 1953. The Avro Vulcan was the world's first large delta wing bomber making its first flight for the Royal Air Force on 3 August 1952. However, the North American Aviation XB-70A Valkyrie, the proposed supersonic bomber prototype, which only flew as a research aircraft, dwarfed the Vulcan's size.

The shape of the delta wing keeps the wings away from the shock waves formed by the nose cone of an aircraft flying at supersonic speed. At the low end of the speed range, the Rogallo wing used for hang gliders is another excellent example of the delta wing.

*See also* **Gothic delta; Ogival/Ogee delta wing; Vortex lift.**



**Density** Density, denoted by the Greek letter ' $\rho$ ' (rho), is defined as the mass per unit volume (mass/volume). To be precise, it should be stated whether it is absolute, static, or relative density that is being considered. Notice mass is used in density calculations, not weight.

The density of air depends on the air temperature, pressure, and humidity (moisture of the air). Air pressure and density are directly proportional, where a decrease in pressure results in a corresponding decrease in density. However, air temperature and density are inversely proportional, where a decrease in temperature results in an increase in air density. Air density decreases with altitude due to the fact air pressure decreases with altitude. Temperature also decreases with altitude, which should increase the density but it is the decrease of air pressure that has the greatest effect. At an altitude of 23000 feet, the air density is half the value of sea level air density, at approximately  $0.592 \text{ kg/m}^3$ . [Note the air pressure is half the sea level value at about 18000 feet].

The density of moist air is slightly lower than dry air at the same temperature and pressure, but can be taken as being the same value by pilots for practical purposes. From the above, it can be seen that density depends on the air temperature and air pressure, from which it is calculated using the equation of state for air – the ideal gas law.

Low air density has an adverse effect on aircraft performance, especially when the temperature is higher than average and/or the altitude is some way above sea level. The result is a high-density altitude, which has an adverse effect on light aircraft taking-off from a mountain airstrip, for example. Low air density is found when the temperature is higher and/or the air pressure is low.

In the aviation industry, density of air is of importance in meteorology, navigation, and aircraft performance and of course, aerodynamics. Density can be measured in various units depending on which branch of science is being considered. The density at sea level is assumed by the International Standard Atmosphere to be:

1.225 kg/m<sup>3</sup>  
0.516 g/cm<sup>3</sup>  
0.002378 slugs/cu.ft  
0.0765 lb/cu.ft

In engineering science, the Imperial unit is one slug per cubic foot, where one slug equals 32.2 pounds per cubic foot.

Galileo Galilee (1564–1642) determined air resistance (drag) is directly proportional to the air density ( $\rho$ ) in which the object (plane) is moving. The term density is from the Latin word *densus*, meaning thick.

Dry air density can be calculated from the formula:

$$\text{Air density kg/m}^3 (\rho) = \frac{P}{RT} = \frac{1013.25}{287.05 \times 288 \text{ K}}$$

Where P = absolute air pressure in hPa

R = specific gas constant = 287.05

T = absolute temperature Kelvin.

**See Absolute density; Density ratio; Standard density; Relative density.**

**Density altitude** Pressure altitude adjusted for non-standard temperature determines the density altitude, which is the height in feet above sea level measured in terms of air density. The combination of pressure altitude and temperature determines the density altitude. Density altitude is the same value as pressure altitude only when the ambient conditions are equal to the International Standard Atmosphere conditions. Invariably, there will be a difference in either pressure altitude, temperature or humidity, where an increase in one or all factors will cause an increase in the density altitude. Humidity has less effect on air density than pressure or temperature.

At higher density altitudes, the aircraft's performance is reduced due to less lift from the wings, less thrust from the propeller (or helicopter rotor) and reduced power from the engine(s).

The reduced engine power causes slower acceleration on take-off, which requires a greater take-off distance and an increased true air speed for the same indicated air speed for rotation. Reduced power reduces the rate of climb after take-off. It follows, air density is an important factor when considering aircraft performance, particularly in relatively higher altitudes, high temperature and high humidity.

*Compare with **Pressure altitude**.*

**Density ratio** The air density ratio is the ratio between sea level air density and the density of air at a chosen altitude, represented by the Greek symbol  $\sigma$  (sigma) where the density ratio is given thus:

$$\text{Density ratio } (\sigma) = \frac{p}{p_o}$$

Where  $p$  = density at altitude

$p_o$  = density at sea level (standard density).

The density ratio decreases from 1.000 at sea level to 0.246 at 40000 feet.

The density ratio is a very important factor in aerodynamics, aircraft performance, and air speed indicator corrections.

*See **Absolute density; Standard density; Relative density**.*

**Departure** An aircraft's un-commanded divergence in any axis, from the required flight path.

**Departure stall** A departure stall occurs during the climb-out after take-off or a go-around when the pilot inputs too high a pitch attitude. Retracting flaps too soon, or an out of trim condition can also contribute to the departure stall.

It is also known as a power-on stall.

**Descending turns** A climbing turn causes an over bank condition. However, a descending turn may or may not cause the aircraft to over bank. Two opposing effects counteract each other, one tending towards over bank and the other to roll the wings towards a level attitude.

During the descending turn, the inner wing descends at a larger angle of attack and travels less distance than the outer wing. The outer wing descends at a smaller angle of attack but operates at a higher speed due to the larger radius (and distance) travelled. Therefore, the over bank tendency caused by the outer wing's higher speed will be counteracted by the inner wing's larger angle of attack. Which effect has the greatest tendency will determine if the aircraft will over bank or roll level. The effect can vary between aircraft types and angle of bank. At low bank angles, lateral stability is more likely to level the wings, however at higher bank angles the tendency is stronger to roll further into the turn.

**Design cruise speed ( $V_C$ )** The design cruise speed is the aircraft's forward speed at which the propeller produces its maximum efficiency.

The aircraft's design cruise speed  $V_C$  also appears on the manoeuvre envelope where it is calculated at 33 times the square root of the wing loading.

*See Pitch/diameter ratio; Blade angle ( $\beta$ ); Blade twist; Fixed-pitch propeller.*

**Design dive speed ( $V_D$ )** Flying the aircraft at speeds between the never exceed speed ( $V_{NE}$ ) and the design dive speed ( $V_D$ ) may cause airframe structural distortion. However, flying above the design dive speed will exceed the airframe's certified loading limits; the aircraft could experience structural failure or an in-flight break-up. The design dive speed is calculated to be at 1.4 times the  $V_C$  design cruise speed.

*See Manoeuvre envelope; Gust envelope.*

**Design lift coefficient** The design lift coefficient is the lift coefficient produced at the aircraft's design cruise speed with the wing operating at the optimum angle of attack; this occurs when the mean camber line is parallel to the freestream flow at the airfoil's leading edge.

**Design load factor** The design load factor represents the 50% safety margin added to the limit load factor. The term design load factor is an alternate name for the ultimate load factor.

*See Ultimate load factor; Diagram 59, Manoeuvre Envelope.*

**Design manoeuvre speed** *See Manoeuvre speed ( $V_A$ ).*

**Design point** A combination of the aircraft's forward speed and propeller RPM at which the propeller provides maximum efficiency.

*Compare with Design cruise speed ( $V_C$ ).*

**Detached shockwave** The detached shockwave (bow wave) is a common occurrence on the wing leading edges and aircraft nose of aircraft flying at supersonic speeds and on hypersonic blunt-nose re-entry vehicles such as the Space Shuttle. The Space Shuttle and orbiting space capsules have a very blunt nose, more akin to a Boeing 747 than a pointed nose as found on Concorde, for example.

The blunt nose is used to advantage on re-entry vehicles re-entering the Earth's atmosphere; the blunt-nose generates a detached bow shockwave for dissipating the large pressure energy heat away from the fuselage. The distance of the shockwave ahead of the body depends on the craft's Mach number, temperature behind the shockwave and the size and shape of the body's nose.

At low supersonic speed, just beyond Mach 1.0, the leading edge or nose will compress the air in front of it and increase the heat,

which in turn increases the local speed of sound. The compression waves will then increase their velocity (they travel faster in higher temperatures) and push the bow wave further ahead of the body. This speed is known as the Mach detachment speed ( $M_{DET}$ ). A condition is eventually reached where the bow shockwave stabilizes some distance ahead of the aircraft's nose, and then leans rearwards with increasing Mach number.

Detached shockwaves are a disadvantage on aircraft flying at low supersonic Mach numbers. Therefore, it follows it is advantageous to have the bow shockwave to remain attached to the leading edge, where the flow compression is reduced and it stabilizes the airflow over the wing. This is the reason supersonic fighter aircraft have small radius (sharp) leading edges on their wings. The name is taken from the bow wave formed by ships at sea.

The discovery of the detached shockwave was made by the American Edmund V. Laitone (1915–2000) in 1947.

*Compare with **Attached shockwave**. See **Diagram 8, Detached Normal Shockwave**.*

**Diameter of turn** To find the diameter of a turn in feet for a given angle of bank, divide the speed in 'Knots squared' by the factor in the table below.

Use the factor given in brackets – it is close enough when flying the aircraft. The radius of turn is commonly quoted in aerodynamic textbooks for airplane hypothetical performance. However, as far as pilots are concerned, the diameter of the turn is of greater importance. When the 180-degree turn is completed the aircraft will be twice the radius distance from the start of the turn – the practical distance.

| Bank angle | Factor            |
|------------|-------------------|
| 20°        | $Kts^2/2.0$ (2)   |
| 30°        | $Kts^2/3.0$ (3)   |
| 40°        | $Kts^2/4.8$ (5)   |
| 50°        | $Kts^2/6.7$ (7)   |
| 60°        | $Kts^2/9.8$ (10)  |
| 70°        | $Kts^2/15.5$ (16) |

*See* **Rate of turn; Radius of turn; Angle of bank ( $\varphi$ ); Turning.**

**Diamond wings** The diamond wing is a futuristic concept of a multi-wing aircraft. The lower/forward main wing is swept aft to join the higher/rear forward sweep wing at the wingtip-mounted fins. From most viewpoints, the wings form a diamond shape, hence their name.

It is also known as a rhomboidal wing or joined wing.

**Differential ailerons** When roll control is applied to produce a turn, the amount of movement of each aileron is different. The unsymmetrical movement amounts to about 30° up-aileron and about 15° down-aileron, hence the name differential ailerons. The up-going aileron moving through a greater arc than the down-going aileron produces greater drag to oppose the adverse yaw.

Differential ailerons were introduced to aircraft in 1936.

*See* **Adverse aileron yaw. Compare with Frise ailerons.**

**Differential pressure** The difference in pressure between two points is the differential pressure.

**Dihedral action/effect** The effect of dihedral is to produce a rolling damping moment to restore the airplane back to straight and level flight following a disturbance in the roll axis. Although it is not in itself rolling stability, it is a contributing factor towards it.

Consider an aircraft flying in turbulence and one wing drops; Due to its stability, the aircraft will briefly sideslip towards the lower wingtip. This will increase the lateral angle of attack on the lower wing and hence, the lift. Conversely, the upper wing will have a reduced angle of attack and less lift. The difference in the lift force on each wing will produce a rolling damping moment and return the wings to a level attitude. It follows, for each degree change in sideslip angle the dihedral effect changes the rolling moment. To be precise, the difference in lift on each wing is due to the sideslip angle – not dihedral effect, although it is a contributing factor.

The vertical centre of gravity also influences the dihedral effect, which is increased with a lower vertical centre of gravity location. However, Dutch roll or roll-yaw coupling can be the result of too much dihedral angle.

*See* **Lateral dynamic stability; Lateral static stability; Rolling damping moment.**

**Dihedral angle** The dihedral angle is the angle between the horizontal and the wing up-tilt towards the wingtips. For reasons of stability, most aircraft have their wingtips raised above the wing roots at a small dihedral angle, which is obvious when viewed from in front of the aircraft. The amount of dihedral angle varies between aircraft types, because an increase in dihedral angle increases the dihedral effect. For example, high wing aircraft due to their inherent pendulum stability have less dihedral than low wing aircraft.

Sir George Cayley first described the dihedral angle is a requirement for lateral stability in 1808–9.

*See* **Lateral dynamic stability; Lateral static stability; Rolling damping moment; Dihedral action/effect.**

*Compare with* **Anhedral.**



**Directional divergence** Directional divergence is an unstable, diverging sideslip condition due to insufficient static directional stability.

An aircraft with static directional stability will right itself from a sideslip or yawing condition. However, if the plane is directionally unstable a moment will be created, which tends to increase the divergence from a sideslip or yaw. If the sideslip is allowed to continue, structural failure could occur due to the side force acting on the aircraft.

**Directional static stability** An aircraft's tail fin is a symmetrical airfoil and under normal straight and level flight conditions, it will be at zero angle of attack and not produce any sideways force in either direction. However, if the aircraft experiences a yaw during flight, the vertical tail will turn to an angle to the oncoming airflow. The fin will then have a positive angle of attack of a few degrees and produce a sideways lifting force – it has static directional stability, which will tend to pull the tail back into the original line of flight.

Centre of gravity variation has little effect (within given limits) on directional stability.

It is also known as weathercock stability.

**Dirty configuration** The aircraft is considered 'dirty' when the flaps/leading edge devices and undercarriage are extended. It is also referred to as being 'cluttered'.

*See* **Configuration**.

**Disc actuator theory** *See* **Momentum theory**.

**Disc area** A helicopter's rotor blade disc area is that circular area enclosed within the circumference of the tip path plane. An increase of power will cause the rotor blades to cone upwards, effectively reducing the disc area by a small amount. It is also known as the propeller disc area when referring to a propeller, which can be considered as a constant area because propellers do not cone.

**Discing** The use of 'ground fine pitch' on constant-speed propellers for braking purposes is known as discing. Ground fine propeller pitch operates in the range from zero degrees to about 5 to 6 degrees. Fixed-shaft turboprop aircraft need to be started in ground fine pitch to avoid propeller drag retarding the engine's spool-up. The propeller will operate in the range between ground fine pitch and flight fine pitch when taxiing. In ground fine pitch, the propeller disc will act as an airbrake during the landing roll. A squat switch on the undercarriage enables the ground fine pitch whenever the aircraft is on the ground.

*See* **Ground fine pitch.**

**Disc loading** The helicopter's all up weight divided by the rotor disc area defines the disc loading measured in pounds/square inch or kilograms/meter<sup>2</sup>. It is the helicopter's equivalent to the airplane's wing loading, or propeller blade loading.

Disc loading increases during manoeuvres such as the landing flare and steep turns, where an increase in lift is required. An increase in disc loading requires more power from the engine to provide extra lift from the rotor blades. The lift formula,  $Lift = C_L \frac{1}{2} \rho V^2 S$ , shows that only the lift coefficient can be increased by increasing the rotor blade's angle of attack. [All other factors are considered to be constant including  $V^2$ , or rotor RPM]. Increasing the angle of attack causes the total reaction to tilt away from the axis of rotation increasing the rotor blade drag and rotor torque.

Therefore, more power is required with increased disc loading to overcome the increased drag and to produce increased rotor thrust and hence, increased lift.

Propeller disc loading is measured by engine power/propeller disc area.

**Displacement thickness** The displacement thickness is an alternate name for the boundary layer. It is dependent on the Reynolds number of the flow.

*See* **Boundary layer; Reynolds number.**

**Dissociation of gas** The hypersonic flow through the detached bow shockwave over a re-entry vehicle causes dissociation (separation) of diatomic gas molecules, namely nitrogen and oxygen (the two main gases in air), which are present in the atmosphere. At higher hypersonic speeds, ionization effects become evident.

**Dissymmetry of lift** Dissymmetry of lift is the uneven distribution of lift over a helicopter's rotor disc caused by the difference in velocity between the advancing and retreating rotor blades. To be precise, the term dissymmetry of rotor thrust should be used, because the problem is caused by the uneven distribution of rotor blade thrust. However, the term dissymmetry of lift is the normally accepted term.

If the helicopter had no flapping hinges, then the advancing rotor blade would produce more lift due to its greater air speed (rotor RPM plus the helicopter's forward speed) than the retreating blade (RPM minus the helicopter's forward speed). This would cause the helicopter to roll side to side and to pitch nose-up due to precession.

The cure to prevent the dissymmetry of lift is to use flapping hinges to allow the individual blades to flap up and down. The advancing blade is allowed to flap upwards where the relative

airflow would then be perceived as coming from above the blade, thus reducing the angle of attack and therefore reducing the lift. The rear moving blade, in turn is allowed to flap down and the relative airflow then rises from below and increase the angle of attack and hence the lift. Therefore, the lift becomes equal on both sides of the rotor disc due to the blades flapping up and down. This is known as flapping to equality.

Retreating blade stall restricts the helicopter's forward speed and is one of the reasons for its published never exceed speed ( $V_{NE}$ ).

The tail rotor also experiences dissymmetry of lift due to the vertically mounted tail rotor's advancing blade having greater speed than its retreating blade. Flapping is a requirement to correct for dissymmetry of lift on the tail rotor, just as it is on the main rotor.

Cierva first employed the flapping hinge method on his Autogiros circa 1922. Prior to this, the development of early helicopters was delayed.

*See Flapping hinge; Cierva, Juan de la.*

**Dive brakes** Early World War II dive-bombers were equipped with dive brakes to enable steep dive-bombing attacks to be made with reasonable accuracy.

Dive brakes are similar in appearance to trailing edge flaps but with many perforations along their length. They were found on aircraft such as the Douglas SBD Dauntless and German Junkers JU-87 Stuka dive bombers (one of the earliest aircraft to employ them). The dive brakes were designed to produce more drag than the maximum drag of a normal flap. Some types of trailing edge flaps could be positioned to dive brake mode by extending past their normal full flap position.

Dive brakes are similar to speed brakes, which can be mounted either above or below the fuselage, where as spoilers/lift dumpers, as found on modern jet transport aircraft, are mounted on the upper surface of the wing only.

*See Air brakes; Spoilers; Speed brakes; Diverters; Dive brakes; Lift dumpers.*



**Photo:** The flaps on the Douglas A-24 Dauntless dive-bomber split up and down to provide very effective dive brakes. The National Museum of the US Air Force In Dayton, USA is home to this example.

**Dive recovery flaps** Dive recovery flaps were installed on the wing's undersurface on WW II, P-38 and P-47 fighter aircraft; their purpose was to alter the trim and the airflow under the wing to assist in recovery from high-speed dives during combat manoeuvres, when compressibility was present and caused the nose to pitch down.

See **Compressibility**.

**Divergence** Divergence has two meanings in the aerodynamic sense. In the first reference, divergence refers to the aircraft's flight path and stability, or lack of it. The second reference to divergence concerns the aeroelasticity of the aircraft's structure.

An aircraft with negative static stability can experience divergence from its original flight path. Divergence due to the stability of the aircraft is unacceptable; the aircraft tends to divert from the attitude selected by the pilot without any stable recovery of its own accord. In the lateral sense, if the aircraft enters a sideslip, roll or yaw it will continue to diverge from the straight and level and if left uncorrected, will enter a steep spiral dive. A divergence in the longitudinal plane will place the aircraft into a nose-dive or a pull-up into a stall condition.

The aircraft's structure can suffer the effects of divergence due to violent aeroelastic instability at high dynamic pressures, when they become greater than the restoring forces due to the structure's aeroelasticity. Structural deformation and failure may occur at the critical failure speed when the aircraft's speed exceeds the design dive speed ( $V_D$ ) and its inherent elasticity is too low. The aircraft's design must ensure aerodynamic forces are not of sufficient strength to overcome aeroelastic restoring moments, throughout the aircraft's permissible speed range. Sweptback wings have the advantage of being in trail, an effect that raises the divergence speed, as opposed to forward sweep wings, which by their nature of leading, lowers the divergence speed.

*See* **Manoeuvre envelope; Wing divergence – aeroelastic.**

**Divergence dynamic pressure** *See* **Divergence speed.**

**Divergence speed** The speed at which a wing will be torn from an aircraft due to aeroelastic deformation following a high-speed overstress is known as the divergence speed.

Above the divergence speed, the aerodynamic forces overpower torsional resistance to the point of structural failure. Divergence speed is set at a speed 50% greater than the aircraft's maximum allowable speed.

It is also known as divergence dynamic pressure.

*See* **Aeroelasticity; Flutter; Flexural aileron flutter; Mass balance; Forward sweep wings; Reversed controls; Divergence.**

**Divergent intake duct** The air intake duct on a fanjet engine as used by jet transport aircraft is a good example of a divergent duct, where the intake opening is of smaller diameter than the compressor/fan behind it. The duct diverges (expands) as it approaches the compressor/fan. The divergent duct induces the intake air velocity to reduce and the kinetic energy is thus converted into increased pressure energy before entering the compressor/fan. This increase in pressure energy is known as ram recovery. The shroud also reduces compressibility effects on the compressor fan.

*See* **Convergent/divergent duct; Convergent duct; Ram effect/recovery; Venturi.**

**Diverter** Diverter are another form of air brake, which split into two sections and swing outboard on their hinges. The BAe 146 jet airliner has diverter air brakes mounted on the rear fuselage tail cone. The Space Shuttle also has diverters – the rudder splits vertically into two separate panels when deployed as air brakes. Diverter were first used on the low-level strike aircraft, the Blackburn Buccaneer, which made its first flight in 1958.

*See* **Air brakes; Spoilers; Speed brakes; Dive brakes; Lift dumpers.**



**Photo:** A Blackburn Buccaneer with the tail cone, diverter speed brakes extended. This plane is on display at the Newark Air Museum, England.

**Dog tooth** A dogtooth, or leading edge notch, is usually located on the inboard leading edge of a sweptback wing, but may also be found on straight wing aircraft. The outer half of the wing has its leading edge extended forwards of the inner portion of the wing. Some early high-performance jet fighter aircraft with sweptback wings utilize a dogtooth leading edge. The Hawker Hunter was one of the first designs to use a dogtooth. The device generates a strong



vortex over the wing's upper surface re-energizing the airflow to improve the swept wing's performance.

Also known as a saw tooth, leading edge notch or rebated leading edge.

**Dorsal fin** The dorsal fin is located on top of the fuselage centreline and merges with the vertical fin. Its purpose is to delay or prevent fin stalling and reduce the yaw moment coefficient; this will increase the longitudinal stability.

The opposite of the dorsal fin is the ventral fin, which is mounted under the fuselage by the tailplane.

**Double-delta** A double-delta wing has a sharp leading edge sweepback angle changing to less sweep angle around the midpoint between the wing root and wingtip.



**Photo:** The Swedish SAAB J35 Draken fighter first flew in 1955 and is an excellent example of a double-delta wing aircraft. It has a maximum speed of 1320 MPH. A Draken is shown here at Newark-on-Trent Air Museum in England.

The Swedish SAAB J-35 Draken fighter is a unique double-delta wing design. The outboard/aft wing section leading edge is swept back at approximately 50 degrees while the inner/forward part of the delta is swept back at approximately 80 degrees, more like a large strake, but all part of the main double-delta wing. The first aircraft fly with a double-delta wing was the 7/10 scale, SAAB 210 Lill-Draken, which was used to test the double-delta concept.

The Russian Tupolev TU-144 'Charger' supersonic airliner also has a double-delta wing with 76 degrees of sweep on the inboard wing section decreasing to 57 degrees on the outboard panels.

It is also known as a compound delta.

See **Delta wing**.

**Double-slotted flap** A flap system with two separate flap vanes separated by two slots, one slot between the wing's trailing edge and the first flap vane and the second slot between the first and second flap vanes.



**Photo:** The double slots are clearly shown on these extended flaps.

**Download** Most aircraft have a download acting on the tailplane in normal straight and level flight to counteract the effect of the forces of lift/weight and thrust/drag. The lift/weight couple tends to pitch the nose down and the thrust/drag couple tends to pitch the nose up. Generally, the lift/weight couple wins out with a greater increment of down pitch. Therefore, a download provided by the tailplane is required to trim out the remaining pitching forces.

See **Stabilizers**.

**Downwash** Due to its shape an aircraft wing in subsonic flight experiences an airflow upwash ahead of the wing and a downwash component off the wing's trailing edge. The airstream velocity increases as it flows over the wing's upper surface, and in order to change the airstream velocity, energy is required, which is obtained from the energy generated to produce wing lift and is then given up as extra induced drag. The downwash, due to its greater velocity has greater effect on lift than does the upwash. Proponents of the downwash theory of lift claim that wing lift is due to the downwash and without it, there would be no lift! In fact, the downwash is directly proportional to the *lift coefficient*, not total lift.

The aspect ratio of the wing is a major factor in the characteristics of the downwash. Low aspect ratio delta wing aircraft for example, operate with a relatively larger angle of attack at low speed producing a high lift coefficient and greater downwash, which in turn forms a larger trailing vortex and associated induced drag. Conversely, a high aspect ratio wings deflects a larger mass of air over the longer wing and therefore the downwash is reduced with a weaker vortex system being produced and hence, less induced drag. The downwash vortex is stronger near the wing tips than at the wing root.

The trailing edge downwash and its associated vortices affect the airflow in several ways:

- it has greater influence on the airflow velocity over the wing's surface
- the geometric angle of attack consists of the section angle of attack plus the induced angle of attack
- the total reaction component of lift is tilted rearwards to remain at right angles to the average relative wind vector due to the modified downwash over the wing
- the horizontal vector of the lift force represents the induced drag
- the downwash angle is measured relative to the plane of symmetry (OY lateral axis) and is directly proportional to the lift coefficient
- a cusp, as found on the rear under surface of a supercritical airfoil, can reduce the downwash, considerably improving wing performance.

The downwash that is forced downwards from helicopter main rotors does provide lift, unlike wing downwash, which is airflow deflected downwards, not forced.

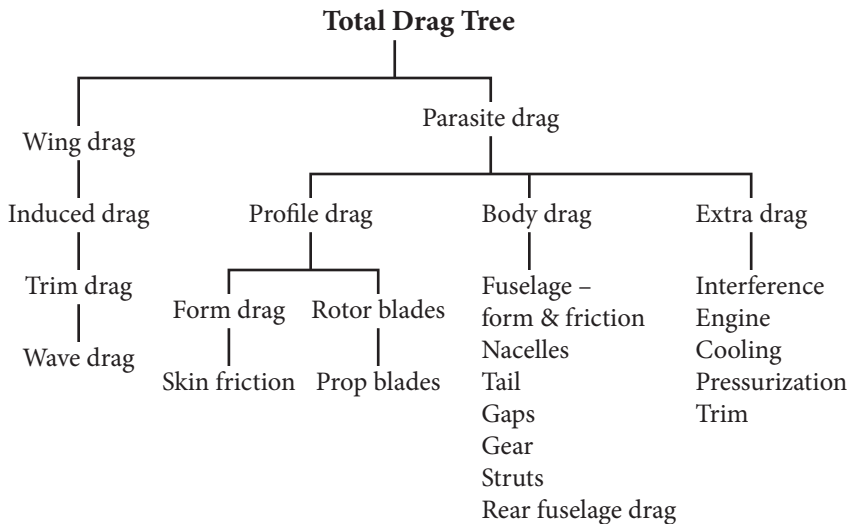
The Biott-Savart law can be used to estimate the extent of the downwash. The downwash angle is given the Greek letter ( $\epsilon$ ) epsilon. The term downwash was introduced in 1915. Induced downwash is an alternate term.

See **Kutta condition; Plane of symmetry; Induced drag; Diagram 43, Induced Drag.**

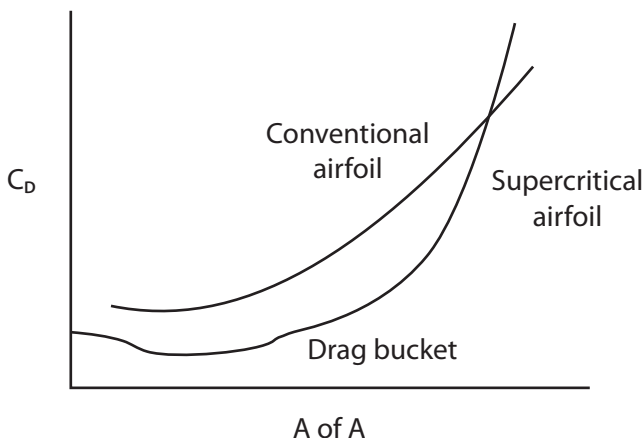
**Drag** Drag, acting in the free-stream direction of flight, is the shear stress (air resistance) of all retarding air particles flowing over the aircraft due to the viscosity of the air causing friction and turbulence in the boundary layer.

The total drag acting on the aircraft is divided into two basic groups of wing drag and parasite drag. Wing drag is classed as induced drag (lift-dependant drag). Parasite drag can be further sub-divided into profile drag, body drag and extra drag. The Total Drag Tree below lists the different types of drag present on an aircraft.

Lord Rayleigh and Jean le Rond d'Alembert were both instrumental in developing drag theories.



**Drag bucket** The term drag bucket refers to a dip in the drag curve, which indicates the angle of attack range where the transition point movement is delayed with increasing nose-up pitch, due to a favourable pressure gradient. The laminar flow maintains its presence over a greater part of the wing chord with the first few degrees increase in angle of attack. The lift coefficient remains constant over this range of angles of attack with the friction drag reduced to form the drag bucket.



**Diagram 18, Drag Bucket**

Aircraft designers may choose a wing with a drag bucket for maximum efficiency covering the aircraft's proposed cruise speed range. Supercritical wings have drag buckets, which are especially suited to modern jet transport aircraft. **Diagram 18, Drag Bucket**, compares the drag curve for a non-supercritical wing and a cambered supercritical wing. The drag bucket range on the graph is also known as the favourable range or favourable cruise speed range. The drag above and below the drag bucket range may be higher than a non-supercritical wing, but that is of less importance to the operation of the aircraft.

It is also known as a laminar low-drag bucket.

**Drag coefficient** The dimensionless drag coefficient is a ratio between two numbers in the same units of area and described as the ratio of total drag pressure to dynamic pressure; it is a function of the angle of attack, Mach and Reynolds numbers and aircraft configuration. The lower the drag coefficient, the less is the drag, which is a major requirement in aircraft design. The reference drag coefficient occurs at zero degrees angle of attack. It is found by dividing the aircraft's total drag by the dynamic pressure ( $\frac{1}{2} \rho V^2 S$ ) using appropriate units, thus:

$$C_D = \frac{\text{total drag}}{\frac{1}{2} \rho V^2 S}$$

Where drag = Newtons or pounds

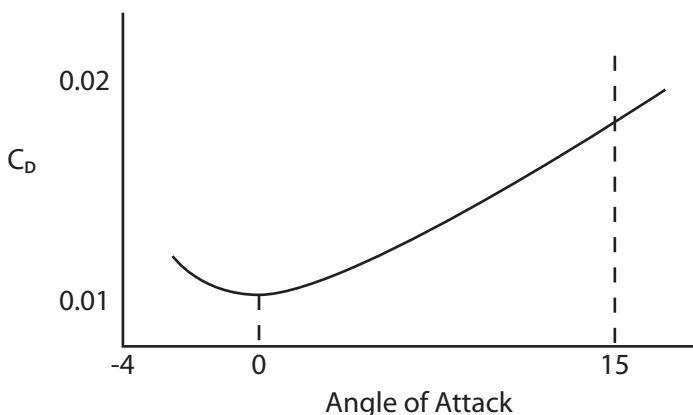
$C_D$  = drag coefficient

$\frac{1}{2}$  = a constant

$\rho$  = air density  $\text{kg/m}^3$  (or  $\text{lbs/cu.ft}$ )

$V^2$  = aircraft speed in  $\text{m/sec}^2$  or  $\text{FPS}^2$

$S$  = wing area in square meters or square feet.



**Diagram 19, Drag Coefficient**

An example of drag coefficients for a few aircraft is as follows; the Cessna 180, 182 and 310 all have drag coefficients of 0.027. In

comparison, the Boeing 747's drag coefficient is slightly higher than that of the Cessna's at 0.031 while the new model, the Boeing 787 is lower at 0.024. The McDonnell F-4 Phantom II supersonic fighter has a drag coefficient of 0.021 when flying subsonic increasing to 0.044 in supersonic flight; drag rise through Mach 1.0 is a reality! The effects of drag or air resistance have been known for some time, as the following shows:

- Leonardo Da Vinci (1452–1519) noted in 1490, drag is proportional to the object's (plane's) area (S) and streamlining reduces it
- Galileo Galilee (1564–1642) determined in 1600, drag is directly proportional to the air density ( $\rho$ ) in which the object is moving
- In 1673, the Frenchman Edme Mariotte (1620–1684) demonstrated drag is proportional to the speed squared ( $V^2$ ).

**Diagram 19, Drag Coefficient**, shows how the drag coefficient is at a minimum value at zero degrees angle of attack for a symmetrical airfoil and increases its value with an increase in angle of attack.

**Drag count** The drag count is just another way to express drag coefficient. A drag count of one is equal to a drag coefficient of 0.0001,  $1/10000$ , or  $10^{-4}$ . An aircraft in cruise flight averages 200 to 400 drag counts. Reducing the drag count by one enables an additional 200 pounds of payload to be carried.

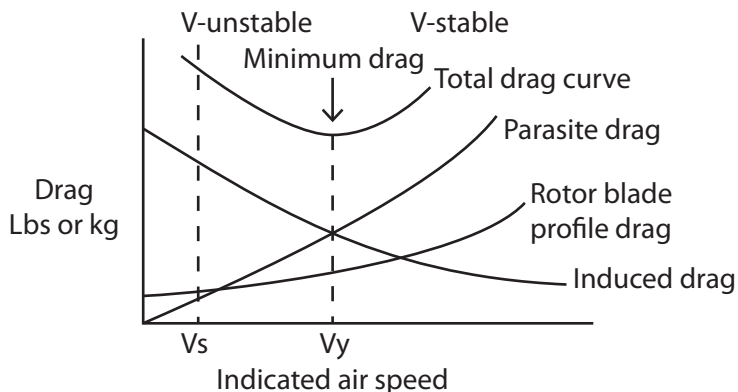
**Drag creep** With increasing Mach number, around Mach 0.8, the drag starts to increase very gradually at first, normally caused by a weak shock wave developing on the wing's upper surface. This is drag creep. Beyond this initial increase in drag, the drag increases more rapidly with increasing Mach number, hence the term drag rise, which is caused by boundary layer separation due to the shock wave increasing in intensity.

*Compare with* **Drag rise**.



**Drag curve** In conjunction with the drag coefficient curve mentioned above is **Diagram 20, Total Drag Curves**, showing the various types of drag for the whole aircraft.

This curve shows the total drag in kilograms (kg) or pounds (lb) against the Indicated Air Speed (IAS). The total drag consists of the parasite and induced drag, plus the rotor blade profile drag on a helicopter. The parasite drag increases in proportion to the velocity squared, while the induced drag decreases inversely proportional to the velocity squared, the velocity being IAS. At approximately 1.3 times the flaps up stall speed  $V_{SI}$ , the parasite drag increase is proportional to the induced drag decrease. Therefore, at 1.3  $V_{SI}$ , parasite, and induced drag are equal, each making up half of the total drag; this occurs at approximately three degrees angle of attack. The maximum lift/drag ratio is also achieved at this point – the  $V_Y$  minimum drag speed.



**Diagram 20, Total Drag Curves**

**Diagram 20, Total Drag Curves**, shows at the stall, the total drag curve rises sharply. Most drag diagrams show a steep climb in the backside of the power curve on the graph, but ignore the effect of the extra drag produced at the stall. In reality, the total drag curve rises almost vertically at the stall, which is not normally taken

into account by the generalized equations. At the high end of the speed range, the parasite drag again increases much more than the normally accepted amount when compressibility and wave drag is present in high-speed flight; the second, diverging line on the graph shows this.

On airplanes, the profile drag is part of the parasite drag. However, on a helicopter, the rotor blade profile drag is considered a separate drag due to the rotor blade's rotation. When the helicopter's forward speed increases, the speed-squared factor (as found in the lift formula) becomes prominent and the blade profile drag increases. Therefore, with increasing forward speed, the increasing parasite drag combined with the decreasing induced drag, will cause the total drag to reduce to a minimum and then increase with increasing helicopter forward speed. This is shown on **Diagram 20, Total Drag Curves**, by the rotor blade profile drag curve. Note also, the indicated air speed line is divided into speed unstable and speed stable.

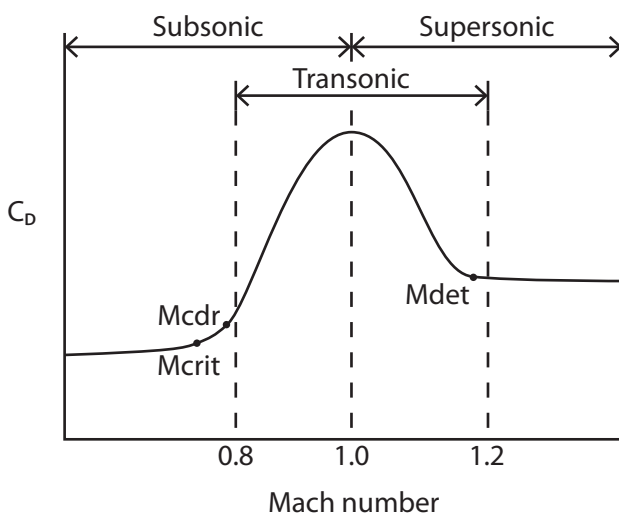
The drag curve is normally plotted against angle of attack. When the drag curve is plotted against lift coefficient, it is termed a drag polar.

*See Speed stable/unstable.*

**Drag divergence curve** The drag of a conventional airfoil will start to increase when the flow velocity reaches approximately Mach 0.7. For a super-critical airfoil, the speed at which the drag increases is approximately Mach 0.8. Before this figure is reached, the airflow is all at subsonic speed. At the critical Mach number ( $M_{\text{CRIT}}$ ), the airflow over the wing will have reached supersonic speed at the wing's point of maximum camber (curvature) and an incipient shockwave will form there and also on other protruding parts of the aircraft.

The value around Mach 0.8, defines the critical drag rise Mach number ( $M_{\text{CDR}}$ ). It occurs at a slightly higher speed than the critical Mach number. Beyond the critical drag rise Mach number,

the drag will continue to rise considerably. Although the boundary layer separation drag is reducing, the energy drag is still increasing. Large areas of supersonic airflow are developing on the wing's upper and lower surfaces to reach a maximum value at the drag hump peak where Mach 1.0 is reached. After the drag hump peak, the drag gradually decreases until at the Mach detachment speed ( $M_{DET}$ ) at approximately Mach 1.2 the drag coefficient is at 1.5 times its subsonic value, depending on the shape of the aircraft's nose and wing leading edges. The Mach detachment speed is the speed at which the bow wave detaches from the wing's leading edge or aircraft nose. Between the values of approximately Mach 0.9 to Mach 1.3, the drag rise is too great for fuel-efficient cruise by any aircraft. Supersonic aircraft must continue to accelerate through this speed region to at least Mach 1.3 or higher for economical cruising. The drag rise or wave drag as it is known, is shown on the drag curve in **Diagram 94, Wave Drag**, where the parasite drag increases with increasing Mach number plus the boundary layer separation drag; this is shown as a hump on the curve.



**Diagram 21, Drag Divergence Curve**

The critical drag rise Mach number ( $M_{CDR}$ ) is used interchangeably with drag-divergence Mach number ( $M_{DIV}$ ), force divergence and drag rise. Less drag will be produced at the drag hump on an aircraft with area rule compared to one without area rule.

Jacob Ackeret (1898–1981) performed the initial investigations into the increase of drag in the transonic region.

Inspection of **Diagram 21, Drag Divergence Curve**, shows the drag rise associated with increasing Mach number at a constant angle of attack and thus constant lift coefficient.

See **Shockwaves; Subsonic flow; Wave drag; Diagram 51, Lift Coefficient & Mach Number**.

**Drag divergence Mach number ( $M_{DIV}$ )** The value (around Mach 0.8) defines the drag-divergence Mach number in the free-stream flow. It occurs at a higher speed than the critical Mach number where a supersonic flow first appears over the wing and compressibility drag starts to increase. **Diagram 21, Drag Divergence Curve**, shows the curve starts to increase rapidly at the drag divergence Mach number; this value is defined as the point where the drag coefficient increases due to an increase in Mach number.

Beyond the drag divergence Mach number, the wave drag continues to increase to eventually peak (at the drag hump) before reducing to a value slightly higher than the subsonic drag value.

Also known as, the critical drag rise Mach number ( $M_{CDR}$ ), critical drag Mach number ( $M_{DR}$ ) and force divergence Mach number ( $M_{FD}$ ).

**Drag equation** The drag equation can be used to calculate the aerodynamic drag for an aircraft using the total drag formula.

Aircraft drag is proportional to the fluid's density and the square of the relative velocity between the body (aircraft) and the surrounding fluid, or air. Surface area also plays its part. Generally, wing area (S) is used, but that can be replaced with frontal area, or the square of the airfoil's chord, which can be transferred to wing area. Wing area has a relatively low drag coefficient when compared to the frontal area, which has a higher drag coefficient than an airfoil with equivalent drag.

The drag coefficient varies with changes in speed ( $V^2$ ), object length thus area (S), air density, and viscosity, collectively known as the Reynolds number. Drag decreases with an increase in altitude due to less air density, shown by  $\rho$  (rho) in the formula.

$$\text{Total drag} = C_D \frac{1}{2} \rho V^2 S$$

Where total drag = Newtons or pounds

$C_D$  = drag coefficient

$\frac{1}{2}$  = a constant

$\rho$  = air density  $\text{kg/m}^3$  (or  $\text{lbs/cu.ft}$ )

$V^2$  = aircraft speed in  $\text{m/sec}^2$  or  $\text{FPS}^2$

S = wing area in square meters or square feet.

The drag equation ( $C_D \frac{1}{2} \rho V^2 S$ ) is attributed to the work of Lord John William Strutt Rayleigh (1842–1919), an English physicist. The French physicist, Edme Mariotte (1620–1684) discovered the fact the speed is squared ( $V^2$ ).

**Drag estimate** An aircraft is a three-dimensional body, and estimating drag is therefore, a rather complex problem. Choices of three-different areas are available to the aerodynamicist to choose for calculating the drag of an aircraft. These areas are the frontal, wetted and wing areas.

See **Frontal area; Wetted area/surface; Wing area.**

**Dragging** Dragging is the angular movement of a helicopter's rotor blade back and forth in the horizontal plane.

The drag hinges also prevent fatigue due to blade bending moments, which could lead to destruction of the rotor system. Drag hinges are found in rotor systems with three or more blades, but not in two blade rotor systems. The slowest blade lags behind the other blades.

Also known as lead-lag.

See **Drag hinge**; **Ground resonance**.

**Drag-grading curve** The drag-grading curve indicates the amount of drag present across the wingspan. The rise in the drag-grading curve at each wingtip is due to the increased induced drag caused by the wingtip vortices. The curve is depressed at mid-span due to the presence of the fuselage and less rake angle between the upper and lower isobars, which indicates the airflow over and under the wing.

See *Diagram 6, Aspect Ratio & Vortex Drag*.

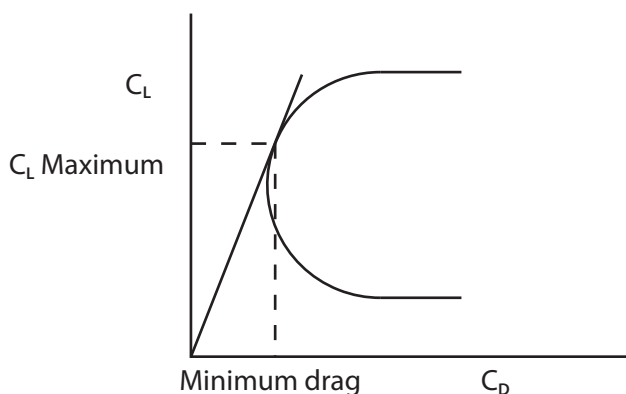
**Drag hinge** Drag hinges are vertical hinges at the blade root of a fully articulated rotor head system on a helicopters. They allow each rotor blade to accelerate or decelerate from their 120° position (on a three rotor system) to prevent violent and destructive oscillations from occurring. They maintain the conservation of momentum.

See **Dragging**; **Ground resonance**.

**Drag hump** Reference to *Diagram 21, Drag Divergence Curve*, shows a drag hump on the drag divergence curve starting beyond the drag divergence Mach number ( $M_{CDR}$ ) and rising to a peak at Mach 1.0, before descending. The drag hump will be higher on the graph for a wing without area rule.

See *Diagram 51, Lift Coefficient & Mach Number*.

**Drag polar** When the drag coefficient is plotted against the lift coefficient, it is termed a drag polar graph. Drag coefficient is plotted on the horizontal X-axis against lift coefficient on the vertical Y-axis. The drag polar shows the drag coefficient's relationship to the lift coefficient as a letter 'C' shaped curve. A tangent line drawn from the origin to intersect the curve will subtend an angle  $\tan \theta$ . A horizontal line from the intersection will show the maximum lift coefficient ( $C_L$ ), and a vertical line will show the minimum drag coefficient ( $C_D$ ) and hence  $C_L/C_D = \text{lift/drag ratio}$ .



**Diagram 22, Drag Polar**

Otto Lilienthal (1848–96) a German pioneer of glider flight introduced the drag polar graph to aeronautics.

**Drag ring** A drag ring is a short/medium chord airfoil around a radial engine designed to reduce drag and improve engine cooling.

See **Townend ring**; **NACA cowling**.

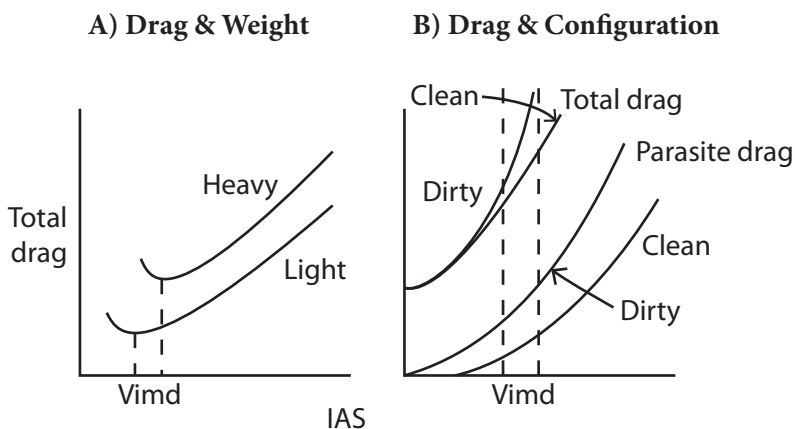
**Drag rise** With increasing Mach number (beyond about Mach 0.8) the drag rises due to boundary layer separation caused by a shock wave on the wing's upper surface increasing in intensity.

*Compare with Drag creep.*

**Drag rise Mach number ( $M_{DR}$ )** See Drag divergence Mach number ( $M_{DIV}$ ); Mach number ( $M$  or  $Ma$ ); Drag curve.

**Drag/velocity ratio** The drag/velocity ratio is the ratio of the aircraft's drag to velocity.

**Drag, weight, & configuration** An increase in aircraft weight will move the total drag curve upwards to the right and the minimum drag ( $V_{IMD}$ ) speed will increase. **Diagram 23A** shows this. Assuming now a constant weight but a change in the aircraft's configuration with the application of drag increasing devices – undercarriage, flaps, spoilers, etc, and parasite drag will



**Diagram 23, Drag, Weight & Configuration**



be increased with no effect on the induced drag. Therefore, the parasite and total drag curves will move up the graph and to the left, as shown by **Diagram 23B**. The minimum drag ( $V_{\text{IMD}}$ ) speed is then reduced: this is an advantage especially for jet transport aircraft, which are very clean aerodynamically and need lower approach speeds.

**Driven region** The driven region is located at the helicopter's main rotor blade tip area, where the RPM is very high and combined with the inflow velocity, the angle of attack is small. This is of significance during an autorotation descent. The total reaction is angled well to the rear away from the axis of rotation, which produces a very poor lift/drag ratio. The rotor thrust reduces and rotor drag increases considerably, which tends to retard the rotor RPM. It is also known as the propeller region.

See **Autorotation – helicopter**.

**Driving region** During a helicopter's autorotative descent, the driving region is a part of a helicopter's rotor blade that covers most of the blade between the blade tip and blade root areas. The driving region creates forward rotor thrust, which propels the blade around to provide lift and rotor thrust during the helicopter's autorotative descent. As found in the stalled root and blade tip areas the inflow comes from below the rotor disc, at such an angle the lift vector is tilted forwards and providing a good amount of lift due to the relatively efficient angle of attack. The drag vector is then acting forwards in the same direction as the rotor thrust which drives the rotor blade. The total reaction is tilted forwards of the axis of rotation.

It is also known as the autorotation region.

See **Diagram 7, Autorotation Forces – Helicopter**.

**Droop** The wing's leading edge is permanently drooped (curved) downwards to increase the camber for greater lift on light aircraft with STOL modifications. The leading edge droop should not be confused with 'droops', which refers to leading edge Krueger flaps.

*See Droops.*

**Drooped ailerons** Drooped ailerons are used as a form of flaperon to create extra lift and drag when lowered for improved take-off and landing performance but retain their ability to act as normal ailerons for roll control. Drooped ailerons are extended in unison with the main flaps but to a lesser angle of deflection. Up to 10–15° of downward deflection is possible with the control column in the neutral position.

Drooped ailerons are not very common, however they are found on a few types of aircraft particularly for STOL application. Some ailerons can also be raised above the neutral position to increase the cruise speed a few extra Knots.

**Drooped leading edge** Drooped leading edges are commonly found on the leading edge of sweptback wings to improve the lift at low-speeds.

The droop leading edge devices are attached to the wing's leading edge and automatically lower when flaps are selected, rotating downwards into the slipstream. By providing a leading edge slot for the air to flow through, the air is re-energized before flowing over the upper surface and delays the boundary layer separation at high angles of attack. A drooped leading edge increases the leading edge radius, which delays the onset of flow separation and hence, a stall at large angle of attack and at a slower speed. They are also known as droops or droop snoots.

*See Leading edge devices (LEX); Krueger flaps.*

**Drooped wingtips** Cessna has employed the use of drooped (or downturned) wingtips on some models of its single-engine fleet to reduce the effects of the wingtip vortices. A few other aircraft types also have used drooped wingtips in the past.

**Droop nose** A droop nose is found on some delta wing aircraft, to provide a better view for the pilots during take-off and landing. It is a prominent feature on the BAC Concorde and the Russian SST Tupolev TU-144 'Charger' for example.

The term droop nose is also applied to a form of leading edge device. The new Airbus A380 jet transport has a solid, hinged leading edge, which is totally sealed without any gaps or slots and droops down with flap application. The droop nose leading edge does the same job as other leading edge devices in improving the airflow over the wing at high angles of attack.



**Photo:** The BAC Concorde is shown here with its nose in the raised position for cruise flight; it droops for take-off and landing. This Concorde is in the Fleet Air Arm Museum in Yeovilton, England.

**Droops** Droops are an alternate name for a leading edge high-lift device, which is hinged and lowers downward to a negative angle in to the airflow to increase the leading edge camber.

*See* **Krueger flaps.**



**Photo:** An Airbus A380 on final landing approach to Auckland, New Zealand.

**Droop snoot** *See* **Drooped leading edge.**

**Dryden Flight Research Centre** The NASA Dryden Flight Research Centre is based on the Edwards Air Force Base, Mojave, California. The research centre was named for Dr. Hugh Latimer Dryden (1898–1965) the Director of NACA from 1947 until the inception of NASA in 1958.

Edwards Air Force Base has been in operation since 1933 and has undergone several name changes over the years. It became the Muroc Army Air Field on 8 December 1943 with another name

change to the High-Speed Flight Research Station in 1949 and is used for NASA's aeronautical high-speed flight-test program for experimental and research aircraft, flight research and atmospheric flight operations. It is the home of supersonic flight for the experimental research X-planes and other prototypes and the Space Shuttle's back-up landing site. Its location was chosen for its perfect clear weather and desert location away from populated areas, ideal for the many speed and altitude records that have been achieved here.

Here at the Dryden Flight Research Centre is where USAF Major Charles (Chuck) Yeager (1923–) became the first man to exceed Mach 1.0 in 1947 when flying the Bell X-1. NACA test pilot A. Scott Crossfield (1921–2006) was the first man to exceed Mach 2.0 in NACA's (later NASA) Douglas X-3 D558-11 Sky Rocket in November 1953.

Edwards Air Force base was renamed from Muroc Army Airfield in honor of Glen Edwards (1918–1947) the test pilot of a Northrop YB-49 prototype who died when it crashed on 8 December 1949.

**Dryden, H. L.** Dr. Hugh Latimer Dryden (1895–1965) was an aerodynamicist and the Director of NACA/NASA from 1946 until his death in 1965. The NASA Hugh L. Dryden Flight Research Centre was named in his honor in 1976.

He is credited with many achievements during his career with NASA, including research into laminar and turbulent airflow, boundary layer problems, wind tunnel design, and development of the North American X-15 rocket plane. This is just a brief outline – his awards and honors list is extensive.

**Drzewieicke, S.** Stefan Drzewieicke (1844–?) further developed the propeller blade element theory from 1892 onwards and he has been credited with the majority of the research work on propeller theory. The initial blade element theory was credited to William Froude (1810–1879).

*See Propeller aerodynamics.*

**Ducted fan** A short radius propeller with an odd number of blades encased in a shroud is known as a ducted fan.

See **Shrouded propeller; Fenestron.**

**Dunne, J. W.** John William Dunne (1875–1949) was an Irish aeronautical engineer, author and a Lieutenant in the British Army. He is noted for being the first person to introduce elevons on a biplane aircraft, which he designed and the first to design swept wing, tailless aircraft. He designed the Dunne D.1, a biplane glider, which was tested in 1907 at Blair Atholl, Scotland. He also designed powered aircraft. The HM Balloon Factory at Farnborough (the forerunner of the Royal aircraft Establishment) built them.

The German, Dr. Adolph Büsemann (1901–1986) is commonly credited with inventing the sweptback wings some several years later.

**Durand, W. F.** William Frederick Durand (1859–1958) is noted for his pioneering work in propeller hydraulics and propeller aerodynamic at the Langley Memorial Aeronautical Laboratory. He published his findings in 1927. He was also a US Naval Officer, pioneer aeronautical engineer, marine engineer and was one of the founding members of NACA and during 1942–46, he became the first civilian chairperson of NACA.

**Dutch roll** Dutch roll is an unpleasant, out of phase, wallowing, short-period oscillation that is insufficiently damped in the yaw, and roll plane.

Yaw is first parameter and roll is second but more noticeable. Dutch roll deteriorates with a decrease in speed and an increase in altitude due to decreased dynamic pressure. Positive directional stability is inherently weaker in the yawing plane than positive lateral stability. A roll will normally induce a sideslip in the direction of roll and the aircraft tends to be restored back to level flight due to strong lateral stability. However, because the directional stability

is weaker than the lateral stability, the restoring yaw motion lags behind the restoring roll motion. The resulting rolling and yawing motion get out of phase causing a self-perpetuating Dutch roll. Aircraft with high wings, dihedral or sweptback wings all have strong, positive lateral stability, which is a desirable asset for all normal flight manoeuvres but it does tend to aggravate the Dutch roll. Therefore, Dutch roll can be stable, neutral, or unstable. It can be counteracted by a yaw damper system, which all high-flying jets are equipped for this reason.

An airplane's Dutch roll motion depends on the ratio of directional stability and dihedral effect; it is related to spiral divergence and directional divergence. Too great a dihedral angle can induce Dutch roll.

The term Dutch roll, now an archaic term in the ice skating sport, was introduced to the aviation world in 1916 by Jerome Clarke Hunsaker (1886–1984) an American aeronautical engineer. It is also known as lateral oscillation.

*See Dihedral action/effect; Lateral dynamic stability; Lateral static stability; Directional static stability.*

**Dynamic heating** The friction of the airflow impacting on forward facing parts of the aircraft in the stagnation regions and the viscous air flow, which is brought to rest by friction on the aircraft's surface, causes dynamic heating.

*See Kinetic heating.*

**Dynamic pressure** Dynamic pressure (or dynamic energy) is the force of air being brought to rest on the stagnation points of the aircraft due to its motion through the air. It represents the kinetic energy of the free airstream flowing over the aircraft. In an incompressible flow, it is the difference between the total pressure and the static pressure, symbolized by 'q', where the dynamic pressure is equal to  $\frac{1}{2} \rho V^2$ .

Dynamic pressure is used in all airflow calculations throughout the speed range from low-speed subsonic flow to hypersonic speed (Mach 5.0+).

The dynamic air pressure has a force (in Imperial units) of 25 PSI at 100 MPH or 100 PSI at 200 MPH varying with the square of the indicated air speed. [Imperial units are used here as an example because the numbers are easier to follow]. Dynamic pressure is measured in pounds per square foot or Newtons per square meter. It contributes to the aerodynamic reaction of the airfoils – that is, lift and drag. It varies with the square of the velocity and is directly proportional to the air density as shown in the formulas below and in the lift and drag formulas.

The air speed indicator measures dynamic pressure to record the aircraft's speed.

$$\text{Dynamic pressure} = q = \frac{1}{2} \rho V^2$$

Where  $q$  = dynamic pressure in Pascal

$\frac{1}{2}$  = constant

$\rho$  = air density  $\text{kg/m}^3$  or  $\text{lbs/cu.ft.}$

$V^2$  = aircraft speed in  $\text{m/sec}^2$  or  $\text{FPS}^2$

$S$  = wing area in square meters or square feet.

It is also known as kinetic pressure or velocity pressure.

**Dynamic similarity** Dynamic similarity is a concept used by aerodynamicists in theoretical, experimental research and computational analysis of aircraft design. Testing a scaled model of an aircraft in a wind tunnel requires mathematical calculations to apply the test results to the full-scale aircraft. The use of the non-dimensional units of lift and drag coefficients and Reynolds number, Mach number, angle of attack are major factors in dynamic similarity. If these given factors from the model match the same values on the full-scale aircraft, then dynamic similarity exists.

*See Reynolds number.*



**Dynamic stability** Dynamic stability is the term used to describe the nature of the plane's positive longitudinal dynamic stability, which returns the aircraft to straight and level flight after several decreasing oscillations. It can be described by the first, second and third modes of damping motion where the resulting motion is damped out with time.

The plane's first mode of long term, phugoid oscillation around the lateral OY axis occurs as it returns to its original trim speed. Depending on the velocity of the aircraft, the resulting sinusoidal wave motion may take between twenty seconds and two minutes or longer, to complete one oscillation as the aircraft seeks to maintain straight and level flight. The time taken for the oscillations to damp out is proportional to the speed of the aircraft. It is acceptable for the aircraft to have positive, neutral, or negative dynamic stability provided the static stability remains positive. The second mode of dynamic stability is a relatively short period restoring moment lasting just a few seconds, which is further reduced due to the pitch damping moment. Third mode occurs during a stick-free very brief oscillation due to heavy damping effect and only lasts for a second or two.

*Compare with* **Stability; Static stability.**

**Dynamic thrust** The term dynamic thrust refers to the thrust available from the propeller that is equal to 'the mass of air moved per second times slipstream velocity minus the aircraft velocity'.

# E

**Effective angle of attack** The effective angle of attack ( $\alpha_{\text{eff}}$ ) is the angle between the local relative wind and the chord line of a two-dimensional airfoil, as would be used in a wind tunnel. It is also known as the infinite aspect ratio angle of attack. The effective angle of attack is  $\alpha_{\text{eff}} = \alpha - \alpha_i$ .

See *Diagram 4, Airfoil Terminology*.

**Effective pitch** A propeller's effective pitch is equal to its advance per revolution, vector B-C on *Diagram 68, Propeller Pitch*; it is related to the helix angle.

**Effective pitch ratio** The effective pitch ratio of the propeller is an alternate name to the advance/diameter ratio. It is used to define a propeller's characteristics.

Also known as the slip function.

**Effective profile drag** The effective profile drag of a given wing is the difference between the induced drag and the total wing drag of an elliptically loaded wing.

See *Elliptical lift distribution*.

**Effective propeller thrust** The net propulsive force supplied by the propeller. The propeller's gross thrust, minus the increased drag of the body behind the propeller.

**Effective span** The wingspan is taken as the total distance from wingtip to wingtip. However, the effective span is this distance minus the correction for tip losses.

**Elevators** The elevators are one of the three primary control surfaces, located on the trailing edge of the horizontal tailplane, which rotate up and down to control the aircraft in the pitching plane around the OY lateral axis.

**Elevons** Elevons are a combination of elevators for pitch control, and ailerons for roll control typically mounted on tailless, delta wing aircraft; Concorde is one example. When used in conjunction, they control the aircraft in pitch and when used in opposition, they provide roll control. Elevons are mounted on the trailing edge of the main, delta wing either side of the fuselage, as opposed to the taileron, which performs the same operation but are mounted as a separate tail unit. The term elevon is derived from the nouns elevator and aileron.



**Photo:** The Convair XF-92A is now on display at the National Museum of the United States Air Force in Dayton, Ohio.

The Irishman, John William Dunne (1880–1949) an aviation pioneer, aeronautical engineer and author invented the elevon circa 1900. It is claimed, Robert Esnault-Pelterie (1881–1957) was the first person to use elevons on his glider in the early part of the 20<sup>th</sup> Century. The American aircraft company Convair introduced the name elevon, when they used elevons on its Convair XF-92 delta wing, which was designed as a fighter aircraft. However, it only saw service as an experimental aircraft.

**Elliptical lift distribution** A perfectly aerodynamically efficient wing will have a spanwise lift distribution, measured in pounds per foot of span, forming a semi-ellipse over the wing's upper surface from wingtip to wingtip. This is the most efficient form of span loading. Calling it an elliptical lift distribution is a misnomer because it is actually a semi-ellipse (the undersurface is ignored).

The induced angle of attack ( $\alpha_i$ ) and the section angle of attack ( $\alpha_o$ ) are both equal throughout the span, as opposed to wings with a non-elliptical lift distribution where the induced angle of attack ( $\alpha_i$ ) and the section angle of attack ( $\alpha_o$ ) both vary along the span. A perfect elliptical lift distribution will induce the downwash to be more evenly distributed across the wingspan reducing the strength of the wingtip vortices and hence induced drag, due to the constant value of lift coefficient to local wing lift coefficient ( $C_l/C_L = 1.0$ ).

It is also known as an elliptical lift-grading curve.

See **Span loading; Induced angle of attack ( $\alpha_i$ )**.

**Elliptical loading** See **Elliptical lift distribution**.

**Elliptical wing planform** An elliptical wing planform is the most aerodynamically efficient type of wing planform. It produces the most exact elliptical lift-grading curve possible, minimizes the induced drag, and reduces stress on the wing. The lift coefficient and downwash is constant from wing root to tip. The tip chord is

reduced to almost zero and therefore, the lift at the wing tip is zero; this improves the aerodynamic efficiency.

However, the elliptical wing is more complex and expensive to build than other shaped wings. Constant chord wings, or wings with aerodynamic or geometric twist are less efficient than an elliptical wing planform and the lift distribution over the wing may be less than a perfect ellipse. The next best compromise is a tapered wing – better than a constant chord wing but not as efficient as an elliptical planform.

Elliptical wings were used on the early US built, Republic P-47 Thunderbolts and more notably on the famous Supermarine Spitfire.

*See Induced drag coefficient; Span efficiency factor.*



**Photo:** This Republic P-47D Thunderbolt resides in the National Museum of the US Air Force in Dayton, Ohio.

**Empennage** The term empennage was an early name for the complete tailplane unit; it includes the vertical fin, horizontal tailplane, rudder, elevators, and trimmers. The name is taken from the French word *empenner*, which means ‘to feather an arrow’.

The empennage is also known as the tailplane.

**End-plates** End-plates are located at or near the tips of the mainplane or horizontal tailplane. They are the forerunner of winglets now commonly found on the tips of the main wings on many aircraft of different types.

They add extra directional stability to the aircraft when mounted on the tailplane. The twin-turboprop Beech 1900D feeder airliner has small end-plates, or fins hanging from its horizontal tailplane. The force on most tailplanes is generally in a downward direction to balance the aircraft longitudinally. The positive pressure and hence, the main lifting side of the tailplane, is on the under surface. The tip vortices generated by the lifting surface, flow from above the upper surface of the horizontal tail to the lower surface – the opposite direction to the main wing’s lift and vortices. Therefore, the small fins below the tailplane act in the same way as winglets to reduce the induced drag and improve the tailplane’s efficiency.

End-plates are also known as end-plate winglets.

Frederick W. Lanchester (1868–1946) a UK engineer, patented the end-plate in 1897. However, in 1929, the aviation pioneer and aeronautics engineer, Vincent Justus Burnelli (1895–1964) is also credited with the invention.

**End plate effects** The aerodynamic effect of a body on the end of an airfoil of finite span, or length.

Wingtip fuel tanks located on the tips of the main wings, and the horizontal tailplane mounted on top of the vertical fin act as end plates. End plates can modify and control the effect of the wingtip vortices and thus reduce the induced drag and improve aircraft performance.

*See Winglets.*

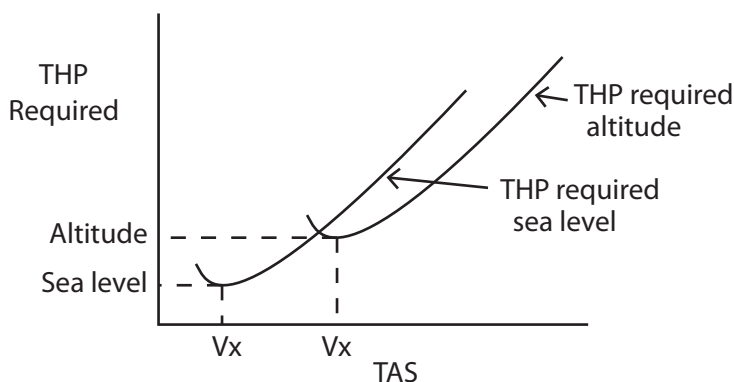
**End plate winglets** See End-plates.

**Endurance** Endurance is the maximum time an aircraft can stay airborne on a given quantity of fuel.

**Endurance speed** The requirement for endurance flying is by definition, to use 'the least amount of fuel per unit time'.

The emphasis here is on time as opposed to velocity used for range flying. Flying for range involves flying at the maximum lift/drag ratio, or minimum drag speed. However, flying for endurance is achieved at the 'minimum power required' speed. The minimum power required speed is always less than the minimum drag speed by almost one third. The speed involved here is the true air speed. At low altitudes, the true air speed will be the same as, or near the indicated air speed. This shows that low altitude is desirable when flying for endurance.

See *Diagram 24, Endurance at Sea level & Altitude; Diagram 60, Thrust & THP Required & Available Curves.*



**Diagram 24, Endurance at Sea level & Altitude**

Because induced drag increases with a decrease in air speed, the drag will be slightly greater at the endurance speed than at the minimum drag speed (range speed). However, the true air speed is also lower with the result, the product of drag times true air speed (power) is also lower. Therefore, less power and more importantly, less fuel is used at the endurance speed than for the minimum drag speed. It follows, if less fuel is used per unit time, the endurance time will be extended for a given quantity of fuel. The difference in endurance between 55% and 75% can amount to an hour or more depending on the engine's fuel consumption. Flying at a lower altitude and with less power will increase endurance. This confirms the statement above 'the altitude required for endurance flying should be as low as possible' consistent with safety, ATC requirements, terrain clearance, etc.

**Energy** The energy in a body describes its capacity to do work. Energy and work are related; therefore, they share the same units of Joules or feet-lbs.

**Energy drag** The term energy drag is an alternate name for wave drag.

*See Wave drag.*

**Energy equation** *See First Law of Thermodynamics.*

**Energy of motion** Also known as kinetic energy.

*See Potential & kinetic energy.*

**Energy of position** Also known as potential energy.

*See Potential & kinetic energy.*



**Energy – potential & kinetic** Potential energy (the energy of position) is the energy that has the potential to do work, hence the name. In other words, it is the energy that has not yet been used (it is stored). For example, potential energy is always present in a descending mass, such as a waterfall, pile driver, etc. [It may be referred to as gravitational potential energy, when acting in the vertical]. Kinetic energy (the energy of motion) is the energy possessed by a moving mass coming to rest or energy that has been or is being used, for example, a hammer blow.

The formula for the potential energy of a falling mass is:

$$PE = mgh$$

When PE is converted to KE, the following can show:

$$(PE) mgh = (KE) \frac{1}{2} mv^2$$

Where  $m$  = mass (kg)

$g$  = acceleration due to gravity ( $9.8 \text{ m/s}^2$ )

$h$  = height (m)

$v^2$  = velocity squared.

Potential and kinetic energy has many applications in the physical world but of interest to us in the aviation world is its relevance to aerodynamics and aircraft performance.

When the aircraft is stationary on the ground the energy is zero. However, when engine power is increased for takes-off, the aircraft accelerates and climbs, and the kinetic and potential energy increases. Kinetic energy is stored in the aircraft's air speed and engine power. Potential energy is gained with an increase in altitude. At the aircraft's maximum straight and level speed, energy is equal to the lift and drag and the speed remains constant, although some energy is lost (transferred) in providing lift and the inevitable drag. Doubling the air speed of an aircraft will quadruple its kinetic energy.

In a descent, some kinetic energy from the engine and some potential energy from the decreasing altitude are again lost with

potential energy being transferred to kinetic energy. Therefore, the total energy depends on air speed, engine power, and altitude. As the plane is flown through all normal manoeuvres, kinetic and potential energy is being used and transferred at all time without the pilot being aware of the fact. This is the conservation of energy.

The important point here is to remember that energy can neither be destroyed nor created but energy is converted to heat, sound and disturbance from air (drag) etc. Also, remember, energy is a scalar quantity not a vector; direction is not considered but speed is.

Bernoulli theorem is related to the potential and kinetic energy of a fluid flow and is stated as 'the dynamic pressure (pressure energy) plus the static pressure (kinetic energy) equals a constant', shown by the equation ( $p + \frac{1}{2} \rho V^2 = \text{constant}$ ). Note the similarities, or should I say differences, in the formulas below for kinetic energy and momentum:

$$\text{Kinetic energy} = \frac{1}{2} mv^2$$

$$\text{Momentum} = mv$$

**Entropy** Entropy (2<sup>nd</sup> Law of Thermodynamics) is a measure of the energy that is unavailable to do work, although the energy still exists. It refers to the disorder of the molecules in a physical system, where greater entropy produces a greater degree of disorder. It can increase but never decrease. Temperature, pressure, and volume are all a function of entropy. The term entropy commonly has the prefix of total or stagnation (entropy).

See **Thermodynamics; Isentropic flow; Isobaric; Adiabatic process; Open cycle; Isothermal.**

**Entropy layer** With increasing Mach numbers the shockwave's entropy also increases, producing a strong entropy gradient, which increases with an increase in shock strength combined with a high vortical flow ensconced in the boundary layer. The entropy layer

is greater near the wing's leading edge or aircraft's nose due to the oblique shock wave. Further aft over the aircraft, the entropy and shock strength reduce due to the shock layer angle reducing to lay almost level with the aircraft's surface.

The normal boundary layer grows in depth due to its close location with the shock layer inducing large velocity gradients in the flow field close to the aircraft's surface; this is known as vorticity interaction.

**Equal and opposite** A term commonly used in aerodynamics where two forces are in opposition and of equal magnitude.

**Equal transit-time fallacy** The equal transit time is a term used where it is assumed the airflow particles, which separate at the wing's leading edge and flow over and below the wing to reach the trailing edge at the same time. It does not, nor does it need to! The equal transit time is a fallacy.

An air parcel separated at the leading edge and flowing over the upper surface travels further due to the concave path length and faster due to changes in the pressure gradient. The lower portion of the air parcel travels a shorter path length compared to the upper surface, with little or no variation in speed, or time. [The air in the lower surface boundary layer is slightly retarded due to skin friction]. Therefore, the split parcels of air do not arrive at the trailing edge at the same time, as the equal transit time suggests; the upper flow arrives before the lower flow.

The equal transit time is associated to the longer path length theory.

**Equilibrium** Equilibrium is the Latin word for 'equal forces' meaning a state of balance with no acceleration.

Equilibrium is achieved when a body (aircraft) is in a perfect state of balance, at rest (static equilibrium) or moving in a straight line, constant velocity motion (dynamic equilibrium) and when the two following conditions are satisfied:

- the sum of the vector forces involved is all equal to zero
- the sum of all clockwise moments are equal to the sum of all anti-clockwise moments, or the sum of all torque forces are zero and the body will not turn or rotate.

A body in equilibrium will obey Newton's 1<sup>st</sup> Law of Motion and remain in steady uniform motion or stay at rest until an outside force affects it. For example, a plane at rest on the ramp experiences a force due to gravity acting vertically downwards. An equal and opposite force acts vertically upwards from the ground acting on the wheels. The net force acting on the plane is therefore zero. Two forces in opposition obey Newton's 3<sup>rd</sup> Law of Motion. A body (aircraft) is therefore in equilibrium when the resultant of all forces acting on the body (aircraft) is zero. The example given above in Newton's laws where lift equals weight and thrust equals drag proves an aircraft is in equilibrium when flying straight and level.

A body (aircraft) is not in equilibrium when it is turning or manoeuvring even though the forces acting on the body are all in balance.

**Equivalent flat plate area** The equivalent flat plate area is a measure of the aircraft's drag efficiency. The flat plate idea is used to represent the parasite drag. The area of a flat plate is normally given the value of 'one'. However, a flat plate placed perpendicular to the airflow records a value of 1.18 due to the air spilling around the edges to become a disturbed air behind the plate. The difference in pressure between the front and rear of the plate is known as the pressure drag. A flat plate with the corrected value is known as the equivalent flat plate area. A streamlined flat plate is then equivalent to an un-streamlined plate of greater area.

The equivalent flat plate area is related to the surface area as shown by the term 'S' in the formula below. It is the ratio of parasite drag to dynamic pressure ( $\frac{1}{2} \rho V^2$ ).

$$\text{Flat plate area (S)} = \frac{\text{parasite drag}}{\frac{1}{2} \rho V^2}$$

The drag equation for a plate =  $C_{DP} \frac{1}{2} \rho V^2 S$

Where  $C_{DP}$  = drag coefficient – flat plate

$\frac{1}{2}$  = a constant

$\rho$  = air density 1.225 kg/m<sup>3</sup> or lbs/cu.ft.

$V^2$  = aircraft speed in m/sec<sup>2</sup> or FPS<sup>2</sup>

$S$  = flat plate surface area in square meters or square feet.

The correct term for flat plate area is equivalent parasite area, but flat plate area is commonly used.

The drag of a flat plate of one square feet at 100 Knots is equal to approximately 29 pounds of force.

*See Flat plate area; Frontal area.*

**Equivalent parasite area** *See Equivalent flat plate area.*

**Equivalent shaft horsepower** The residual exhaust from some turboprop engines also imparts a certain amount of jet thrust from 15% to 25% of the total thrust produced by the engine and when this is added to the shaft horsepower (SHP) available from the propeller shaft, it is then termed equivalent shaft horsepower (ESHP). This is shown by the formula as:

$$\text{ESHP} + \text{SHP} = \frac{TV}{325\eta_p}$$

Where ESHP = equivalent shaft horsepower

SHP = shaft horsepower

$T$  = jet thrust, lbs

$V$  = true air speed in Knots

$\eta_p$  = propeller efficiency.

**Euler's equation** Leonhard Euler (1707–1783) developed his well-known momentum equation in 1752, as an equation for inviscid flow for fluid energy and conservation of mass. The equation compares the values of density and internal energy at various points in the flow field, to find pressure energy. The equation is significant in the study of vorticity in transonic flow, propeller, and helicopter rotor blade aerodynamics. The effects of viscosity of the fluid was ignored by Euler but later addressed in the Navier-Stokes equation.

The Euler's equation mentioned above, is one of several equations that he devised, which are also referred to Euler's equations but are related to other subjects.

**Euler, L.** Leonhard Euler (1707–1783) a Swiss national lived in the same town of Basel, Switzerland as Daniel Bernoulli (1700–1782) and was tutored by Daniel Bernoulli in mathematics. Three years later Euler received his degree in philosophy. He spent most of his working life in St. Petersburg in Russia and Berlin, Germany.

Applying his knowledge of mathematics and physics, Euler was instrumental in developing the use of mathematics to solve problems in physics. He also developed his differential equation related to pressure and velocity – known now as the Euler equation (there are other equations from Euler) and he was the first person to publish the continuity equation. He also worked alongside Daniel Bernoulli in developing Bernoulli's equation for which he receives partial credit, although it is not recognized as such very often.

**Excess thrust horsepower** Excess thrust horsepower is the power required by the aircraft to climb, over and above that required for straight and level flight.

The thrust horsepower required curve on *Diagram 60, Thrust & THP Required & Available Curves*, is related to the familiar drag curve but plotted in terms of power required after being calculated from the value of drag times velocity and plotted for various

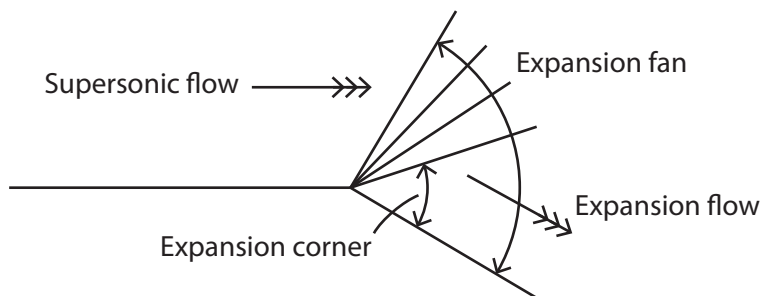
true air speeds. Power required is the minimum power required to maintain straight and level flight. Extra power is required for climbing, turning, and other manoeuvres, known as the excess thrust horsepower. The formula for excess thrust horsepower is as follows:

$$\text{ESHP} = \frac{\text{weight} \times \text{rate of climb (FPM)}}{33000}$$

Compare with **Deficit thrust horsepower; Diagram 60, Thrust & THP Required & Available Curves.**

**Expansion flow & fan** A supersonic flow rounding an expansive (convex) corner forms an expansion fan and becomes a negative expansion flow.

A subsonic flow can easily flow around a corner but a supersonic flow cannot follow the same path and be unaffected. Following the convex corner, the supersonic flow must change direction in stages, which it does by means of an expansion fan. Each individual expansion wave in the fan turns the flow by a small amount. On exiting the fan, the air temperature, density and pressure have all



**Diagram 25, Expansion Flow & Fan**

decreased while the velocity of the flow has increased; the change is opposite to that through a shockwave. It is a gradual change in conditions as opposed to the sudden change through a shockwave.

The expansion fan is not a shockwave as such, but a series of expansion waves (weak shockwaves) each located at the sharp corner. Each expansion wave is at an angle to the surface, with each successive expansion wave angle increasing as the flow progresses around the corner. Due to the increased velocity at the expansion fan's exit, the final Mach line is angled to the flow condition beyond the expansion fan at a steeper angle relative to the surface than the first Mach line. **Diagram 25, Expansion Flow & Fan**, shows this.

An expansion fan is also known as the Prandtl-Meyer expansion fan, continuous or centred expansion wave.

See **Mach angle & cone; Mach lines; Compression flow**.

**Expansion wave** An expansion wave is a weak shockwave within an expansion flow emanating from a point at increasing angles. The expansion wave theory was introduced in 1908 by Professor Ludwig Prandtl (1875–1953) in association with his supersonic shock wave research.

An expansion wave is also known as a rarefaction wave.

See **Expansion flow & fan**.

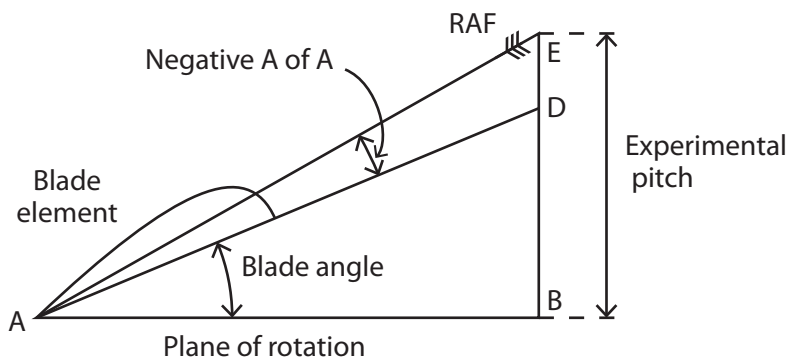
**Experimental pitch** The propeller's experimental pitch is equal to the advance per revolution when the propeller produces zero net thrust.

An inspection of **Diagram 26, Experimental Pitch**, shows as the advance per rev (APR) increases from B to E, the angle of attack will reduce to a negative angle (AD-AE) of around minus two degrees and the propeller will cease to produce thrust. In this condition the relative airflow acts along the line from E to A, and corresponds to the wing's 'zero lift line', to become the prop's zero thrust line. This is the important aerodynamic feature



of the experimental pitch. From the designer's point of view, it is considered as being the 'ideal pitch'. Because it has a definite value and length, depending on the prop's characteristics, it can be used for experimental measurements, hence the name.

A fixed-pitch propeller has only one experimental pitch, while a constant-speed prop's pitch is variable over the available operating range of the blade angles between the fine and course pitch stops. The experimental pitch may also be known as the 'zero thrust pitch' or the 'exponential mean pitch'.



**Diagram 26, Experimental Pitch**

Compare with **Diagram 68, Propeller Pitch** and **Diagram 37, Geometric Pitch** and note the angle of attack and relative airflow vector. See **Advance per rev; Line of zero lift**.

**Exponential mean pitch** The exponential mean pitch is an alternative name for the propeller's experimental pitch.

**Exposed wing area** That part of the wing, which is visible from the wing root to the wing tip, is known as the exposed wing area, or the net wing area. It is used in the calculation of skin friction drag.

*See* **Wing area.**

**Extended propeller blade root** For improved aerodynamic performance and reduced noise from the propeller, the blade roots are enlarged, or extended.

*See* **Blade cuffs; Paddle blade.**

**Extended propeller shafts** Most propeller-powered aircraft have the propeller mounted directly on the engine's drive shaft, which works just fine. However, on some aircraft, the engine cowling causes a disturbance to the passing propeller slipstream, which reacts back on the propeller as interference. This reduces the apparent propeller pitch and its efficiency. To reduce the interference problem the propeller is mounted further forward than normal on an extended propeller shaft. The Piper PA-23 Aztec is one such aircraft that comes to mind with extended propeller shafts.

**Extension flap** *See* **Flaps; Fowler flaps; Area increasing flaps.**

**Extra drag** Extra drag includes all the remaining items listed on the drag tree under extra drag namely, interference, engine cooling, pressurization, and trim drag.

*See* **Drag.**

# F

**Fairey, Sir Charles** Sir Charles Richard Fairey (1887–1957) was an English aircraft designer and manufacturer. He established the Fairey Aircraft Company in 1915 to build aircraft under license to Short Brothers and Sopwith aircraft and after WWI, aircraft of his own design. To name a few of his more famous design are the Swordfish, Albacore, Gannet, Rotodyne and also the first British aircraft to exceed 1000 MPH, the Fairey FD-2. He also developed trailing edge flaps circa 1915.



**Photo:** Here is just one example of the many successful aircraft types, which were designed by Sir Charles Fairey. This Fairey Albacore is on display at the Fleet Air Arm Museum at Yeovilton, England.

**Fairing** A fairing is a tapered, streamlined secondary structure designed to cover drag-producing parts of the aircraft, such as the wheels or bracing struts, etc.

Fairings on the wheels were known as wheel pants (USA term) or trousers, now an archaic term.

**Farman, H.** Henry Farman (1874–1958) was an Englishman born in France. In April 1909, he introduced the aileron for roll control by hinging separate control surfaces, working in opposition on the trailing edge of the four main wings of his Farman *III* biplane. This proved far more suitable for roll control and made the wing warping method patented by the Wright brothers obsolete.

**Farnborough** The Royal Aircraft Establishment's aeronautical research centre is commonly known as Farnborough, located in England.

*See Royal Aircraft Establishment, Farnborough.*

**Fathers of aerodynamics** *See Aerodynamicists.*

**Favourable pressure gradient** A decreasing pressure gradient is known as a negative or favourable pressure gradient. A favourable pressure gradient induces laminar air to flow over the wing with little friction drag. An increase in the pressure gradient induces a negative pressure coefficient and hence lift is produced in the region where the streamlines over the upper surface are curved. However, if the pressure gradient is too small, a loss in lift can be expected.

It is a normal requirement to maintain a laminar boundary layer over as much of the wing as possible to provide the required lift. From the leading edge to the transition point, the velocity increases and the pressure decreases (from high pressure to lower pressure) over the wing's upper surface prompting a laminar flow that adheres to the surface. After the transition point, the flow enters the region of adverse pressure gradient.

*See Diagram 1, Adverse and Favourable Pressure Gradients.*

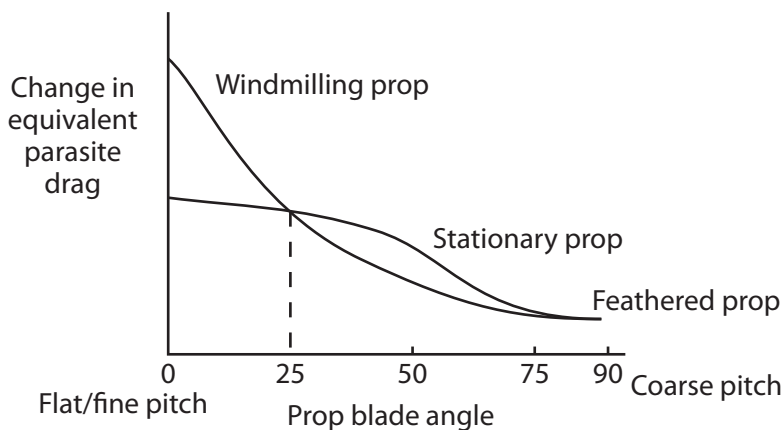


**Photo:** The CASA 235 light turboprop transport is shown here with its propellers in the feathered position, which is normal on stationary free-shaft turbine engines. Note, the streamlined wheel fairings.

**Feathered propeller** A feathered propeller has its blades turned to a pitch angle of approximately  $90^\circ$  edge on to the airflow to reduce parasite drag and vibration caused by the slipstream as it flows over the wing and tailplane. Because of the propeller blade's twist, only the middle portion of the blade is actually parallel to the airflow, while the blade's inner and outer portions are presented to the airflow at a positive angle of attack in opposing directions. The outer portion of the blade will tend to rotate the propeller in one direction while the inner portion will tend to rotate it in the opposite direction. The net result is, both forces cancel out and the propeller remains stationary. Feathering is essential to reduce the parasite drag of a rotating propeller on a failed engine: the drag of a failed and rotating propeller is equal to a flat plate of the same area.

**Diagram 27, Propeller Drag**, shows a large amount of parasite drag is present when the propeller is windmilling in fine pitch at a low blade angle. When the blade angle is increased towards the 90° position, the drag reduces considerably, to be at a minimum when the propeller is feathered. The dotted line shows initially a stationary propeller produces far less drag than a rotating propeller up to about 20–25° blade angle where the drag is about the same as that produced by a windmilling propeller. Increasing the blade angle close to the 90° feathered position will slow the propeller down to a stop, as shown by the two lines on the graph joining.

See **Diagram 95, Windmilling Propeller Forces** and associated text.



**Diagram 27, Propeller Drag**

**Feathering** The feathering action changes the cyclic pitch angle of a helicopter’s rotor blades by rotating them around the spanwise feathering axis. It is akin to a constant-speed propeller’s ability to change blade pitch angle.

See **Cyclic feathering**.

**Feathering axis** The feathering axis is located on a straight line between the blade's root and its tip and is the angle about which the helicopter's rotor blades vary their pitch angle.

The aerodynamic centre and the centre of pressure are located on the blade's feathering axis and the centre of gravity is located close to it.

It is also known as the spanwise axis.

**Feathering pitch** The propeller's feathering pitch angle is selected for stopping the propeller in flight to reduce windmilling drag.

*See Feathered propeller.*

**Fence** A fence is located on the upper surface of the wing as a raised metal strip, in line with the chordwise airflow over the



**Photo:** The MiG 21 PF fighter has a prominent fence. The RAF Museum Cosford, UK, displays this aircraft.

wing. Its purpose is to prevent a spanwise airflow from developing and increasing induced drag at the wingtips, especially over swept-back wings at high cruise speeds.

A spanwise flow can disrupt the airflow near the wingtips due to a thickening of the boundary layer, flow separation and a reduction in the maximum lift coefficient leading to wingtip stalling and possible loss of control at high angles of attack.

With a fence installed, the opposite effects occur. The spanwise flow outboard of the fence is reduced. Between the wing root and the fence, the boundary layer is re-energized and the maximum lift coefficient over the whole wing is increased. By reducing the spanwise airflow, the wing's pitch up tendency is reduced.

Fences, were first used on German military aircraft in WW II and are now common on modern fighter and jet transport aircraft.

Also known as, a flow fence, boundary layer fence or stall fence.

**Fenestron** A Fenestron™ is a helicopter's anti-torque tail rotor mounted in a duct within the tail boom for improved aerodynamic performance, replacing the conventional tail rotor. The word fenestron is from the French 'little window' itself from the Latin *fenestra* meaning 'window'.

The Fenestron's dissymmetry of lift is reduced due to the enclosed shroud protecting the rotor from the normally present disturbed airflow. A Fenestron commonly has an odd number of blades – up to eighteen in total, not fixed as in a normal tail rotor, but free to feather (similar to lead-lag hinges on a main rotor disc). Allowing the blades to feather reduces vibration and noise due to the uneven blade spacing, which distributes the noise over different frequencies. The high number of blades, its location within the tail boom and its high rotational velocity allows a smaller diameter disc than a conventional tail rotor. The fin area surrounding the Fenestron rotor may be unsymmetrical (bulging on one side similar to a cambered surface) to help create a torque force – due to the Coanda effect – to oppose main rotor torque.



Other advantages of mounting the Fenestron inside the tail boom are:

- reduced foreign object damage (FOD)
- reduced drag
- reduced tip losses
- improved anti-torque control
- reduced engine power required in forward flight
- or more thrust developed by more blades of smaller radius
- some protection to any personnel during ground operations
- tail rotor ground strikes prevented.

The disadvantages are:

- reduced tail rotor control in cross-wind operations
- increased drag from the shroud
- increased weight, construction and purchase costs
- determines size of helicopter limit.



**Photo:** The Eurocopter EC-130-B4 is one helicopter type with a Fenestron.

Sud Aviation's SA 340 second experimental prototype helicopter introduced the Fenestron in the 1960s. It is also known as a Fantail, shrouded tail rotor, or nicknamed a fan-in-fin. Following Sud Aviation's introduction of the Fenestron on their SA 340 helicopter, a Boeing/Sikorsky CH-47B was flight tested by NASA with a Fenestron, or Fantail™ as it is known in America. The Fantail was to be used on the cancelled Boeing/Sikorsky RAF-66 Comanche helicopter.

See **Coanda effect**.

**Fillets** Fillets are convex corners found at the intersection of the fuselage and the wing roots on a low wing aircraft.

Their purpose is to blend the two conflicting airflows over the wing root and fuselage into the smoothest flow possible to reduce interference drag and thus increase the flow velocity, especially near the wing's trailing edge. Caltech, in California and the



**Photo:** The Curtiss C-46D Commando was one of the earliest planes to have wing root fillets to reduce interference drag. This model is on display at the Pima Air & Space Museum in Tucson, AR.

National Physical Laboratory in London, England, both addressed the problem of airflow interference at the wing roots in 1931. Interestingly, some models of the Pfalz World War I fighter aircraft had fillets.

See **Onglet**.

**Fin** The fin, which may be vertical or v-tailed, is a stabilizer to which the rudder is attached. A fin may also be located on the wingtips of the main planes. The fin's purpose is to provide directional stability. The fin area may be increased by adding a dorsal fin to improve stability further, if required.

See **Longitudinal static stability**.

**Fineness ratio** The fineness ratio of a streamlined body, for example a fuselage, airfoil, nose cone, or engine nacelle is a measure of its thickness (or diameter in the case of the fuselage) as a percentage of the length. It is a comparison of its length 'a' to its depth 'b' i.e., the a/b ratio or fineness ratio. A ratio of 15% is average for a well-streamlined body.

See ***Diagram 4, Airfoil Terminology***.

**Fine pitch** Fine pitch refers to the pitch angle of a constant speed propeller's small blade angle of 2–3° used for take-off and landing.

It is also known as flat pitch.

**Finite wing** A wing of fixed span with wingtips, as found on any aircraft is a finite wing. In wind tunnel experiments, a finite wing gives more accurate experimental data than an infinite wing.

**Finite-wing theory** The term finite-wing theory concerns the aerodynamic experimental process carried out in wind tunnels as opposed to infinite-wing theory.

*See* **Infinite wing.**

**First Law of Thermodynamics** The 1<sup>st</sup> Law of Thermodynamics deals with the conservation of energy balance; energy is neither created nor destroyed but can change form. It states the principle of the conservation of energy and mass are related to the transfer of heat entering or leaving a system as a form of energy. The rate of heat added and the rate of work done on a system determines the rate of change of energy in the system. Simply, this is the energy equation which states, heat plus work equals energy change; This can be achieved by an adiabatic, isentropic or reversible process.

The first law also introduces the term of entropy.

*See* **Entropy; Thermodynamics; Second Law of Thermodynamics.**

**Fishtail shockwaves** A supersonic flow over the wings in excess of Mach 1.0 will induce the shockwaves above and below the wing to move rearwards to meet at the trailing edge. Their angles will be inclined rearwards to form a V-shaped shockwave similar in appearance to a fishtail. A subsonic airflow will be formed behind the shockwaves, which will eventually become a supersonic flow as the aircraft's speed increases greater than Mach 2.0.

*See* **Shockwaves.**

**Fixed-pitch propeller** The simple type of propeller with non-adjustable blade angles as found on low performance light aircraft.

The advantages of the fixed-pitch propeller are its simplicity of operation for low-time pilots. It is the cheapest type to install on an aircraft and is relatively maintenance free due to the absence of a constant-speed unit. Its disadvantage is, it give its maximum

efficiency at only one air speed, known as the design air speed. At any other speed above or below the design air speed, the efficiency of the propeller will be reduced.

**Fixed slat** A fixed portion of the wing's leading edge forming a permanent slot directs airflow over the wing's leading edge at high angles of attack. Early aircraft had fixed slats prior to retractable slats becoming more efficient and popular.

*Compare with Slots; Slats.*

**Flap angle** The angle between the flap's chord line and the wing, usually measured in degrees, or sometimes in 'notches'.

**Flap-back** Flap-back of the rotor disc can occur when a helicopter moves forward in flight.

The rotor disc tilts down at the front to produce a forward acceleration of the helicopter. However, the advancing blade becomes the fastest moving blade, producing the most lift when the blade is abeam the helicopter. Due to gyroscopic precession acting at  $90^\circ$  in the direction of rotation, the blade on arriving at the front of the helicopter attempts to rise. The opposite effect occurs on the retreating blade – it descends or flaps back. The total rotor thrust is then inclined rearwards, which has the effect of retarding the forward motion, the fuselage reacts by swinging nose upwards due to pendulum action, and the helicopter could come to a stop. This requires a further control input by the pilot – full forward cyclic control – to lower the front of the rotor disc to keep the helicopter accelerating forwards. Full forward cyclic to keep the nose down limits the helicopter's forward speed. Therefore, when the helicopter's speed increases, flapping eliminates the dissymmetry of lift but can introduce the problem of flap-back.

It is also known as blowback.

**Flaperons** A flaperon combines two separate control surfaces – flaps and ailerons – into one control surface. They are used as ailerons for roll control and as flaps in the normal sense. The advantage is achieved by drooping the ailerons to use as flaps, which has the effect of increasing the flap area.

The Royal Aircraft Factory in the UK experimented with flaperons in 1914 on their short-lived S.E.4 biplane and the Fairey Aviation Company used them from 1916, but it was not until after World War II that they became more common. Today, they are used on a variety of aircraft (for example, the General Dynamics F-16 Fighting Falcon) and in particular, aircraft designed for STOL performance.



**Photo:** The General Dynamics F-16 Fighting Falcon is equipped with flaperons. Photographed in the National Museum of the US Air Force, Dayton, Ohio.

**Flapping** The action of flapping is the vertical rise and fall of a helicopter's rotor blade as it oscillates up and down on the delta or flapping hinge due to the inertia and centrifugal forces of the blades and changes in lift force as the blades rotate.

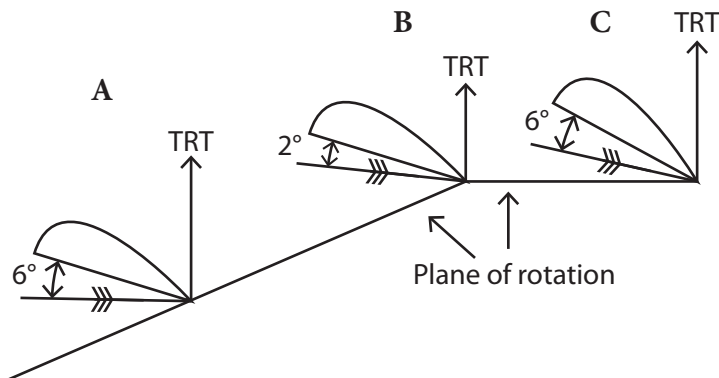
See **Blade flapping**.

**Flapping hinge** A helicopter's rotor blade system utilizes flapping hinges to allow the blades to move up or down (flapping) as they rotate. This action equalizes the lift across the rotor disc (dissymmetry of lift) due to the difference in speed between the advancing and retreating blades. The hinges are mounted on the blades with their axis parallel to the plane of rotation. The two basic types of flapping hinges employed in rotor systems are the Delta-3 and the offset flapping hinge.

It is claimed, the Frenchman, Captain Charles Renard (1847 – 1905) was the first person to invent the flapping hinge for helicopter rotor blades in 1904. However, Juan de la Cierva (1895 – 1936) re-invented the flapping hinge circa 1922 and received the credit for doing so. The hinge allowed the advancement of the helicopter, as we know it today.

See **Flapping to equality; Delta-3 hinge; Offset flapping hinge.**

**Flapping to equality** A helicopter's rotor blades are designed to flap up and down on their flapping hinges when rotating to equalize the lift over the rotor disc with cyclic stick control inputs.



**Diagram 28, Flapping to Equality**

Consider when a helicopter moves forward from a hover; due to inertia, the rotor blade continues along its original plane of rotation with the same relative airflow (*Diagram 28A*). Because the blade pitch has decreased, say to two degrees, (*Diagram 28B*) the total rotor thrust will also decrease and the blade flaps down (*Diagram 28C*). The plane of rotation is now inclined forward from its original position and with the pitch angle remaining the same, the angle of attack increases back to its original value. Likewise, the reverse happens when the blade experiences an increase in pitch, as it does when it reaches the opposite side of the disc. The blade will have then flapped to equality.

**Flaps** Wing flaps are secondary flight control surfaces located on the trailing edge of the main wing between the fuselage and ailerons.

The main purpose of flaps is to increase the boundary layer energy and the maximum lift coefficient to lower the stalling, take-off and landing speeds. With flap application, the pressure distribution over the upper surface changes and increases the loading on the aft portion of the wing.

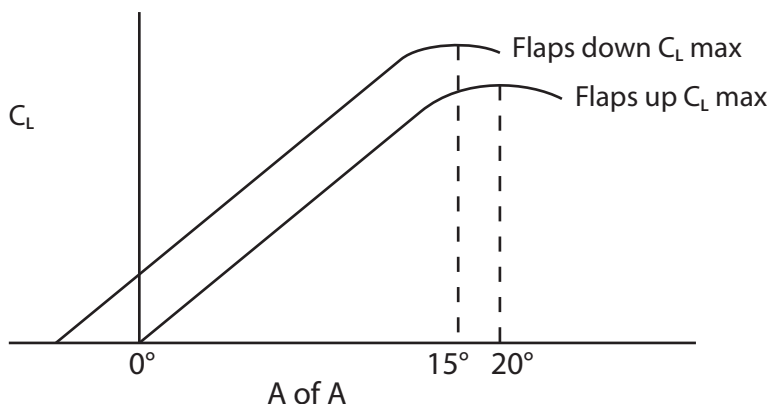
With the flaps deployed, the lift coefficient is increased for all angles of attack, and the lift coefficient curve moves up the graph and to the left. A comparison of the different types of flaps as follows:

- plain (or hinged) flaps increase the maximum lift coefficient from approximately 1.3 up to 2.4, a 51% increase in maximum lift, the critical angle is about  $15^{\circ}$
- leading edge slot and slotted flaps, increase the maximum lift coefficient to about 2.9, an 80% increase in maximum lift and a critical angle of about  $19^{\circ}$
- a split flap increases the maximum coefficient to about 2.6, an increase of 17% in maximum lift and a critical angle of about  $14^{\circ}$



- A Fowler flaps increase the lift coefficient up to about 3.5 or more, an increase in maximum lift of 90% and a critical angle of about  $15^\circ$ .

The second purpose of flaps is to increase the form drag to shorten the landing distance and to increase the angle of approach when landing to provide better terrain clearance, allow the pilot a better view of the runway, and improve speed control. The critical angle is decreased by two or three degrees with full flaps deployed.



**Diagram 29, Lift Coefficient, Flaps Up, Flaps Down**

Increasing the lift and drag coefficients affects the lift/drag ratio; the drag increases more so than lift at greater flap deflections. Flaps increase the wing camber or curvature of the upper surface of the wing. Extending the flaps rearwards and/or deflecting downward (Fowler flaps) increases the chord, and the wing area. Wind tunnel research at Langley Memorial Aeronautical Laboratory pointed out the need for flaps on aircraft to control the landing speed and descent angle.

Reference to **Diagram 60, Thrust & THP Required & Available Curves**, shows the difference in thrust horsepower (THP) required and thrust horsepower available. Lowering the flaps increases the

drag and THP required, shown by the line moving up the graph, which reduces the excess thrust horsepower (ETHP) and increases the deficit thrust horsepower (DTHP), which is also shown by **Diagram 17, Deficit Thrust Horsepower v. KTAS**. The result is, the rate of climb is reduced and the rate of descent is increased with the flaps down.

Flap deployment moves the centre of pressure rearwards along the wing's chord causing a change in the aircraft's longitudinal pitching moment, for two reasons:

- with flaps down the lift is increased over a greater amount of the rear portion of the wing, resulting in an increase in downwash velocity and angle of flow. The centre of pressure moving rearwards increases the lift/weight couple, which causes a greater nose-down pitching moment
- the tailplane of a low wing aircraft being located above the main wing extended-chord line is less affected by the main wing's downwash. However, the tailplane of a high wing aircraft is placed lower than the main wing's extended-chord line and therefore, it experiences the full influence of the main wing's downwash. On a high wing aircraft, the effect of the downwash on the tailplane is further increased causing the nose to pitch upwards on initial flap application, although it may pitch down with further flap application.

Conversely, raising the flaps after take-off moves the centre of pressure forward with a nose-up pitch change. An increase in angle of attack is required to maintain the same lift coefficient, while the drag coefficient decreases allowing the air speed to increase. The final result on which way the nose pitches is a variable depending on the aircraft type.

Louis Bleriot was the first person to use flaps on a powered aircraft when he installed them on his Bleriot № VIII monoplane in 1908. Orville Wright (1871–1948) and James M. Jacobs invented the split flap in 1920 and Handley-Page invented his slotted flap circa 1920, while the German Dr. Hugo Junkers (1859–1935) also

introduced his single and double-slotted flaps in 1921. These flaps were followed in 1924 when the American Harlan Davey Fowler (1895–1982) invented the Fowler flaps, commonly used on most jet transport aircraft today. Trailing edge flaps became more popular and essential in the 1930s when the faster and more streamlined monoplanes replaced the slower biplanes.

*See* **Area increasing flaps; Fowler flaps; Krueger flaps; Lift augmentation devices; Slotted flaps; Split flaps; Leading edge devices (LEX).**

**Flap track fairings** Flap track fairings are mounted on the trailing edge of an aircraft wing – more so on large jet transports.

Flap track fairings cover the flap track mechanism to streamline the airflow to reduce interference drag. They also have the added



**Photo:** The flap track fairings are shown here on the McDonnell Douglas Globemaster III in the retracted position. The fixed (right-hand) forward portion and the moveable, (left-hand) aft portion is clearly visible.

bonus of improving the airflow over the wing to which they are attached. The fairings usually extend past the flap's trailing edge to further reduce wave drag and improve the wing's rear loading on some aircraft types. The fairings are made as two separate parts; the forward section is non-moving and fixed to the wing's trailing edge and the aft section is attached to the trailing edge flaps and moves with them.

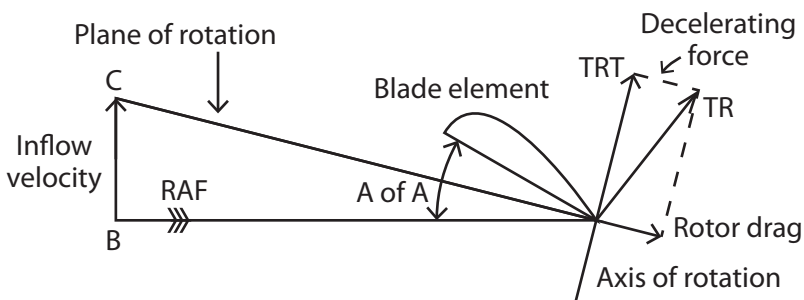
On early transport aircraft, they were built as quite large appendages but tend to be smaller on modern aircraft; they act in a similar way to area rule.

Flap track fairings are also known as anti-shock bodies, speed bumps or pods, canoe fairings or Küchemann carrots, after their inventor Dr. Dietrich Küchemann (1911–1976).

**Flap vortices** Vortices emanate from the flaps outboard trailing edge when extended, due to the greater pressure differential over the extended flaps compared to the relatively lesser pressure differential over the wingtips. Under the right atmospheric conditions, the central core of the vortex will become visible trailing from the flaps but not visible from the wingtip vortices.

**Flare** The final stage of flight where the aircraft transitions from the approach to the landing attitude just before touchdown. In the flare, the aircraft's nose is pitched upwards a few degrees to increase the angle of attack as the speed reduces. With increasing angle of attack, the induced drag increases while all other drag components decrease with decreasing speed.

To pilots, it is known as being 'in the flare'.



**Diagram 30, Flare Effect – Helicopter**

**Flare effect** When the helicopter is in the flare and about to touchdown, the rotor disc is tilted rearwards. Rotor drag reduces forward speed as the helicopter comes to a hover due to the rearward tilt of the total reaction aiding the effect of the decreasing parasite drag. Reference to ***Diagram 30, Flare Effect – Helicopter***, shows the plane of rotation (A-C) is tilted nose-up and the relative airflow approaches from below the disc during the flare as opposed to its normal approach from above the disc. This is due to the induced inflow velocity (B-C) flowing through the rotor disc from below. A greater nose up attitude (more reverse tilt) increases the deceleration force.

**Flat Annular wing** The annular wing is flat and circular with a hole in the centre – a closed wing.

The annular wing design precedes WWI, and although the design was surprisingly stable, the concept never gained popular favour with the pilot population. It is also known as an annular wing or ring wing.



**Photo:** The replica Lee-Richards Annular Biplane has an unusual circular wing planform. This aircraft is located at the Newark Air Museum, UK.

**Flat pitch** Flat pitch is an alternate name for a propeller's fine pitch. The Americans favour the term flat pitch.

*See Fine pitch.*

**Flat plate area** The correct term for flat plate area is the equivalent flat plate area, but the term flat plate area is acceptable.

*See* **Equivalent flat plate area; Frontal area.**

**Flat spin** An aircraft is in a flat spin when the fuselage is in a near horizontal attitude and rotating mainly about the OZ normal or vertical axis. The OX longitudinal axis is almost horizontal as the aircraft descends. The angle of attack increases to about 50° and the rolling movement has mostly disappeared to be replaced by mainly a yaw movement. The wing on the inside of the spin develops a reverse flow aiding the autorotation to continue. The tail fin is in a stalled condition and elevators, rudder, and ailerons are all ineffective in controlling the aircraft.

The flat spin is the worst type of spin an aircraft can enter; recovery is more difficult than a normal nose-down spin. Getting the nose to drop below the horizon is the first requisite for any type of spin recovery. This maybe impossible for some types of aircraft in a flat spin. Twin-engine aircraft are more prone to flat spins than are their single-engine counterparts, due to the weight of the engines on the wings adding to the spin's centrifugal force.

*See* **Inverted spin; Normal spin; Precision spin; Spinning.**

**Flettner, A.** Anton Flettner (1885–1961) was a German aviation engineer and inventor who invented the servo and anti-servo tabs and also the control tab named after him – the Flettner tab. In 1926, he formed his own company where he carried out pioneer work on autogyros and helicopters and introduced the first helicopter (synchropter) designs to have intermeshing main rotor blades – the Flettner FL-265. He moved to the USA after the end of WW II, where he became the chief designer of the Kaman Company. He later formed a company of his own in 1949, in New York.

**Flettner tabs** The Flettner tab is hinged to the rear of the control surface and it moves in opposition to it via cables. This allows the pilot to move a control surface (usually the elevator) with less pounds of pull. Anton Flettner invented it during World War I. It is also known as a servo tab.

*See* **Balance tab; Spring tab; Trimming devices; Geared tabs.**

**Flexible wing** Dr. F. M. Rogallo (1912–2009) invented the flexible wing in 1948 now commonly used by hang glider and microlight pilots the world over. It is more commonly known as the Rogallo wing.

The flexible wing when used on a microlight consists of a lightweight framework of aluminum tubing and formers, which holds the wing to the required aerodynamic shape during flight. The pod fuselage containing the occupants and engine is attached to the wing's hang point through the monopole. The two-axis control system (pitch and roll – no yaw) is controlled through weight shift



**Photo:** The Quantum 912 microlight makes use of a Rogallo flexible wing.



where the control bar moves the pod relative to the wing, which displaces the centre of gravity producing opposite movement around the craft's axis when compared to a conventional aircraft. Push the control forward and the nose rises and vice-versa and moving the control bar to the right will turn the craft to the left, which can be confusing for a pilot of conventional aircraft flying a microlight for the first time. Adverse yaw in a flexible wing craft is prevented by the wing's inherent increase in form drag on the inside wing, which assists the craft to turn.

Microlights using flexible wings are also known as flex-wings, flexiwings, or trikes.

*See Rogallo, Dr. F. M..*

**Flexiwing** *See Flexible wing.*

**Flexural aileron flutter** This is due to the bending of the wing across the span where the outer portions of the wing oscillate up and down, quite visibly in larger aircraft when flying in turbulence. The wing flexing may induce aileron flutter, also known as buzz.

It is caused by the aileron's centre of gravity being located behind the aileron's hinge line. When the wing flexes up and down, the aileron lags behind due to inertia and changes the wing camber rapidly in the wingtip area increasing and decreasing the lift. Flexural aileron flutter is then intensified and can be catastrophic if allowed to occur. Careful aircraft design can prevent its occurrence.

*Compare with Torsional aileron flutter.*

*See Aileron reversal; Flutter; Mass balance; Forward sweep wings; Divergence; Aeroelasticity; Divergence speed.*

**Flexural washout** A high aspect ratio wing, such as a swept-back wing may flex during flight. It is then possible for the wing's leading edge to deflect downwards, more so near to the wingtips. This has the effect of producing washout, a reduction in the wing's angle of attack and thus lift over the outer portion of the wing.

*See* **Flutter; Aileron reversal; Wash-out.**

**Flight envelope** The aircraft's flight envelope is determined by the design of the aircraft type and defines the limits of the aircraft's operational and aerodynamic performance, weight limits, air speed, and altitude, rate of climb and power, etc. The envelope is gradually increased through certification flight-testing with every parameter being tested to prove the aircraft's safety and that it performs as or better than expected.

The term flight envelope is also given to a diagram of aircraft speed versus height. The term 'pushing the envelope' refers to operating the aircraft very close to its performance limits.

It is also known as the manoeuvre envelope.

**Flight research facilities** *See* **Aeronautical research facilities.**

**Flow attachment** Flow attachment is an alternate name for the **Boundary layer**.

**Flow fence** *See* **Fence.**

**Flow line** A flow line represents the average path taken by fluid elements in a fluid flow.

*See* **Streamline.**

**Flow separation** On a wing's upper surface boundary layer, the viscosity of the fluid may cause flow separation, which in turn, causes pressure drag, and changes the airflow from laminar to a turbulent flow. The point where this occurs is the separation point/region located at the front of the separation bubble. If the boundary layer flow reattaches to the surface aft of the bubble, it is known as the point of reattachment.

The Karman vortex street phenomenon is a form of flow separation.

**Flow strip** *See Stall strip.*

**Fluid dynamics** Fluid dynamics is the study of fluid (including airflow) in motion and its interaction with a body (aircraft).

The properties of temperature, density, pressure and velocity all play their own interrelated parts in fluid dynamics, which is related to aerodynamic drag and jet engine thrust, for example.

In the 18<sup>th</sup> Century, Daniel Bernoulli (1700–1782) and Leonard Euler (1707–1783) both studied fluid dynamics including the relationship between momentum, force and acceleration, etc. Jean d'Alembert (1717–1783) also made great contributions to fluid dynamics, laying down the foundation of theoretical fluid dynamics. Bernoulli was also instrumental in introducing the general momentum, energy, and continuity equations – the basis of aerodynamics, which can be applied to the airflow over an aircraft in flight.

It is associated to fluid mechanics and is an early name for classical aerodynamics.

*See* **General momentum equation; Continuity equation.**

**Fluid elements** Small parcels of air known as fluid elements are found in fluid motion.

*See* **Streamline.**

**Fluid flow** Fluid flow is a term common to aerodynamics, which involves the interaction between a body (aircraft) and the surrounding air at subsonic, supersonic, and hypersonic speeds. The various types of fluid flow include:

- **Laminar flow** and **Turbulent flow**
- **Inviscid flow** and **Viscous flow**
- **Compression flow** and **Incompressible flow**
- **Free-molecular flow** and **Continuum flow**.

The study of fluid flow is important to the understanding of aerodynamics and can be subdivided into the above categories or combinations of the above list. For example, an inviscid, incompressible, subsonic flow.

*See separate entries of the above list.*

**Fluid mechanics** The study of fluids and the forces acting on fluids is known as fluid mechanics.

Fluid mechanics can be divided into three groups of:

- fluid statics (hydrostatics) is the study of fluid at rest as found in a closed hydraulic system
- fluid kinematics (hydrodynamics) is the study of a liquid fluid in motion
- fluid dynamics (aerodynamics) is the study of gases (air) in motion over a body (aircraft).

**Flutter** Flutter is a high-frequency vibration in parts of the aircraft structure, usually affecting the control surfaces, wing, or tailplane at relatively high operating air speeds.

The combined effects of aeroelastic forces, inertia forces, and aerodynamic loads on the aircraft's structure cause bending or distortion, which vibrates at high frequency. Changes in angle of attack and its associated pressure distribution also contribute to a resonating oscillation or flutter. If the aircraft's structure has insufficient stiffness, a vibration of high frequency can occur and

lead to total destruction of the aircraft. Flutter can be reduced with the use of static mass balance ahead of the control surface's hinge line, high natural vibration frequency and built-in design strength to ensure the flutter speed occurs well above the normal operating speeds of the aircraft. Increased stiffness of the structure, use of powered controls and avoiding flight beyond the maximum permissible speed will prevent the occurrence of flutter.

With the advent of ever-increasing cruise speeds during the early 1920s several accidents were caused due to flutter. This led to a study of flutter problems and further research in the UK in 1925 by English born Herman Glauert (1892–1934) and Dr. Theodore Theodorsen (1897–1978) of NACA Langley in 1935.

*See* **Aeroelasticity; Flexural aileron flutter; Mass balance; Forward sweep wings; Reversed controls; Divergence; Divergence speed.**

**Flutter speed** The flutter speed is the lowest equivalent air speed (EAS) at which an aircraft will experience **Flutter**.

**Fly-by-wire FBW** An electromechanical signaling system of flight control replacing conventional hydraulic or pushrod and wire cables. The digital fly-by-wire control system enables aircraft to be built with greater manoeuvrability due to less inbuilt inherent stability. The multiple on-board computers over-ride the aircraft's instability to provide precise flight control.

Avro using their Avro 707 swept wing research aircraft to test the idea investigated an early form of fly-by-wire in England starting in 1960. UK's Boulton Paul Aircraft Ltd. followed this, using the second prototype Vickers Viscount 663 (the world's first turboprop airliner) with twin Rolls Royce Tay turbojet engines in place of its usual four turboprop engines, to further test the fly-by-wire concept. The test aircraft was based in 1956, at RAF Seighford near Stafford, UK. The Short SC.1 was the world's first fly-by-wire

VTOL experimental aircraft making its first flight on 26 May 1958 at RAE Boscombe Down, UK.

NASA Dryden Flight Research Centre's famed research project engineer and test pilot, Bruce Peterson (1933–2006) flying the Northrop M2-F2 carried out further research into the fly-by-wire system. An LTV (Vought) F-8C Crusader jet fighter was also used to further flight-test a digital fly-by-wire computerized flight control system, with a first flight in 1972. The North American A-5 Vigilante became the first production aircraft to fly with a fly-by-wire control system, making its first flight on 31 August 1958. The digital fly-by-wire control system eventually found its way to jet transport aircraft via its use on military fighter aircraft, such as the General Dynamics F-16 Fighting Falcon, the Rockwell International Space Shuttle, and the BAC/Aerospatiale Concorde (1969). The French Airbus A320 was the first subsonic jet transport to be equipped with the digital fly-by-wire flight controls with its first flight on 22 February 1987.



**Photo:** The North American Aviation Vigilante, shown here at the Pima Air & Space Museum at Tucson, Arizona, was the first production aircraft to be built with a fly-by-wire control system.

**Flying controls** The primary flying controls of an aircraft are the **Ailerons**, **Elevators**, and **Rudder**. Also included are **Elevons** and **Ruddervator**.

**Flying quality** The relationship between an aircraft's controllability and stability is referred to as its flying quality.

NACA engineer, Dr. Robert Rowe Gilruth (1913–2000) is credited with flying quality theory and test flying research during 1941–43. The Cooper-Harper rating scale is one system now used to define the relationship between good and poor flying quality.

*See* **Cooper-Harper rating scale**.

**Flying tab control** *See* **Spring tab**; **Flettner tabs**.

**Flying wing** A flying wing is an aircraft with a sweptback wing and no fuselage.

John (Jack) Knudsen Northrop (1895–1981) first conceived the idea of an aircraft without a fuselage during the 1930s. He designed the experimental flying wing Northrop XB-35 piston-engine bomber and the YB-49A jet bomber for the US Air Force, but they never went into production. The whole aircraft consists of a wing carrying fuel and payload, etc. Devoid of any external excrescence, drag is cut to a minimum with the whole aircraft producing lift. Longitudinal stability is achieved by using a sweptback wing and tail fin.

The Northrop XB-35 prototype piston-engine bomber made its first flight on 25 June 1946 at Edwards AFB, California. The XB-35 was converted with eight jet engines to become the Northrop YB-49 prototype, which made its first flight on 21 October 1947. These prototype bombers were intended for the USAF but never went into production. Test pilot Glen Edwards (1918–1947) lost his life in the crash of the YB-49 in 1947; Edwards Air force Base was named in his honor.

The English aircraft company, Armstrong Whitworth built a flying wing 1/3 scale model glider during WW II, known as the Armstrong Whitworth A.W. 52G (G for glider). The war's end terminated the glider project. On 13 November 1947, the first of two prototypes A.W. 52s for evaluation by the Royal Aircraft Establishment at Farnborough made its first flight. Two Rolls Royce Nene jet engines powered the first aircraft and the second prototype, powered by two Rolls Royce Derwent engines was operational from 1948 to 1954. John Lloyd designed the two Armstrong Whitworth A.W.52 flying wing aircraft.



**Photo:** This Northrop B-2 Spirit is a modern day return to the flying wing principle. It is now resident at the National Museum of the US Air Force in Dayton, USA.



The German-built Gotha Go 229 was designed by the famed Horton brothers and was the world's first flying wing aircraft to be powered by jet engines, making its first flight in January 1945.

The flying wing concept lay dormant for several years before being revived, not by Northrop but by Lockheed, when they built the Lockheed F-117A Night Hawk stealth strike aircraft. The Night Hawk made its first flight in June 1981 and entered service with the USAF in October 1983. The Northrop Grumman B-2 Spirit stealth bomber, first flight on 12 July 1989, which entered service with the USAF in December 1993, followed the Lockheed Night Hawk. It has the same wingspan as the YB-49.

**Focused sonic boom** An aircraft during a climb or descent at supersonic speed produces audible sonic booms. The series of booms are heard at a given location with each boom travelling at the same speed coming from the aircraft, which is accelerating or decelerating.

*See* **Sonic boom.**

**Folding wings** An aircraft's main wings may be folded upwards or swung back alongside the fuselage.

Folding wings are commonly found on naval aircraft due to the small confines of the hanger decks on-board aircraft carriers. Some light aircraft are also equipped with folding wings for storage purposes.



**Photo:** The Chance Vought F4U-4 Corsair with wings folded and displayed at the Pima Air & Space Museum, Tucson, AR.

**Fore-body inlet** *See* Centre body cone inlet.

**Force** Sir Isaac Newton (1642–1727) described mathematically the meaning of force with the advent of his Laws of Motion. Force (F) is simply a push or pull on a body and is directly proportional to the mass times acceleration. This is shown by the formula:

$$F = ma$$

Where F = Newtons

m = mass (kg)

a =  $m/s^2$

If one unit of force acts on one unit of mass, it will be equal to one unit of acceleration. The SI unit of force is the Newton, symbol N. One Newton is the force required to accelerate a mass of 1 kg at  $1 m/s^2$  in the direction of the force. In Imperial units, force is measured in pound force (lbf) where one lbf will accelerate 1 lb at  $32.2 ft/s^2$ . For very small units of pressure the 'dyne' is used, which is a unit of force required to accelerate a mass of 1 gram at  $1 cm/s^2$ . The conversion factor between Metric and Imperial units is:

$$1 \text{ lbf} = 4.44822 \text{ N} \quad 1 \text{ N} = 0.22480 \text{ lbf}$$

$$1 \text{ kgf} = 9.80665 \text{ N} \quad 1 \text{ N} = 0.10197 \text{ kgf}$$

The acceleration is inversely proportional to the mass for any given force, and is directly proportional to the force. The greater the force the greater will be the acceleration. A force can change momentum of a body, because force is a vector quantity, (it involves both direction and magnitude) the net forces or resultant is the sum of all the forces acting on the body. One horsepower is equal to a force moving 550 pounds over a distance of 1 foot in 1 second or equivalent to a force moving 1 pound over a distance 550 feet in 1 second. This can be related to the work done by an aircraft's engine in moving the aircraft through the air:

Where, 550 feet per second = 325 Knots. Therefore, 1 pound of thrust equals 1 HP at 325 Knots.

The forces acting on an aircraft in flight can be divided into three main groups of aerodynamic (lift and drag, etc) gravitational (weight) and propulsive (engine thrust). These three forces are coupled and interact with each other during flight.

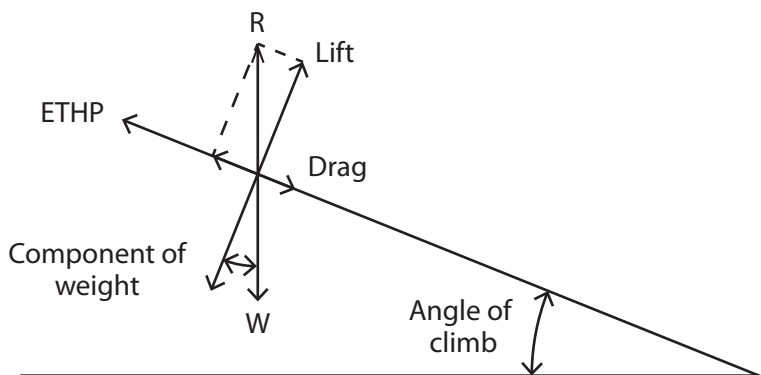
The units for force are the kilogram force (kgf), pound force (lbf) or the Newton.

See **Newton's First Law of Motion; Newton's Second Law of Motion; Newton's Third Law of motion.**

**Force and moment coefficient** See **Coefficients.**

**Force vector** Force vectors are straight lines on diagrams and represent the direction and magnitude of a given force.

**Forces in a climb** Reference to *Diagram 31, Forces in a Climb*, below shows the various forces acting on an aircraft as it climbs to altitude.



**Diagram 31, Forces in a Climb**

During the climb, the lift force is always less than the weight. This may sound confusing to the student new to aerodynamics

where it would be thought the lift should be greater. However, it is not excess lift that enables a plane to climb; it is excess thrust horsepower (ETHP) from the engine. Just as lift is less than the weight, we find thrust is greater than the drag during the climb; the steeper the climb the greater must be the thrust. To explain this apparent anomaly, consider first the lift formula:

$$\text{Lift} = C_L \frac{1}{2} \rho V^2 S.$$

Where  $C_L$  = lift coefficient

$\frac{1}{2}$  = a constant

$\rho$  = air density kg/m<sup>3</sup> or lbs/cu.ft.

$V^2$  = aircraft speed in m/sec<sup>2</sup> or FPS<sup>2</sup>

$S$  = wing area in square meters or square feet.

The lift coefficient is the same in the climb as for straight and level flight at about three degrees angle of attack. Assume the air density ( $\frac{1}{2} \rho$ ) and wing area ( $S$ ) remains unaltered during the climb. This leaves the speed ( $V^2$ ), which is always less than cruise speed during the climb. Therefore, total lift in the climb is less than in straight and level flight.

The vector forces on the diagram shows that thrust equals drag plus weight multiplied by the sine of the angle of climb ( $\gamma$  gamma) and the excess thrust horsepower equals the component of weight acting along the flight path (this is in effect, extra drag). Therefore, total drag equals aircraft drag plus weight sin angle. The resultant ( $R$ ), which is greater than the lift, equals the weight.

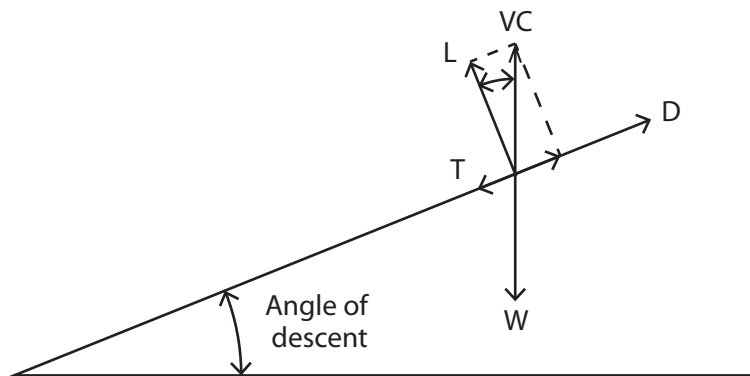
Excess thrust horsepower provides the extra power required for climbing at a given indicated air speed. The vertical component of thrust added to the wing lift equals the weight. As the aircraft climbs to altitude, the excess thrust horsepower gradually diminishes. When there is no longer any excess thrust horsepower available, the aircraft's rate of climb will be reduced to zero feet per minute.

See **Absolute aerodynamic ceiling; Lift equation.**

**Forces in a descent** Reference to *Diagram 32, Forces in a Descent*, shows the relevant forces acting on the aircraft. It can be seen that the vertical component of lift (total reaction) equals the weight, not the lift vector. Engine thrust is negligible assuming the engine is throttled back and so the forward acting component of the weight vector (T) opposes the aircraft drag

The aerodynamic design of the aircraft dictates its lift/drag ratio and so, the greater the ratio the flatter is the glide angle. A decrease in the lift drag ratio, by lowering the flaps, will steepen the glide angle.

Increasing the weight of a given aircraft does not affect its lift/drag ratio. This is because the increased weight is balanced by increased lift and the drag is balanced by an increase in thrust, represented by the 'T' vector. Conversely, when gliding with the aircraft at a lower weight, less lift is required. It follows, both lift and drag being vectors of the aerodynamic total reaction produced by the wings are both reduced at lower weights. By trigonometry, the Tan of the glide angle between the horizontal and the glide path is the same as the angle between the lift force and the total reaction (VC). For

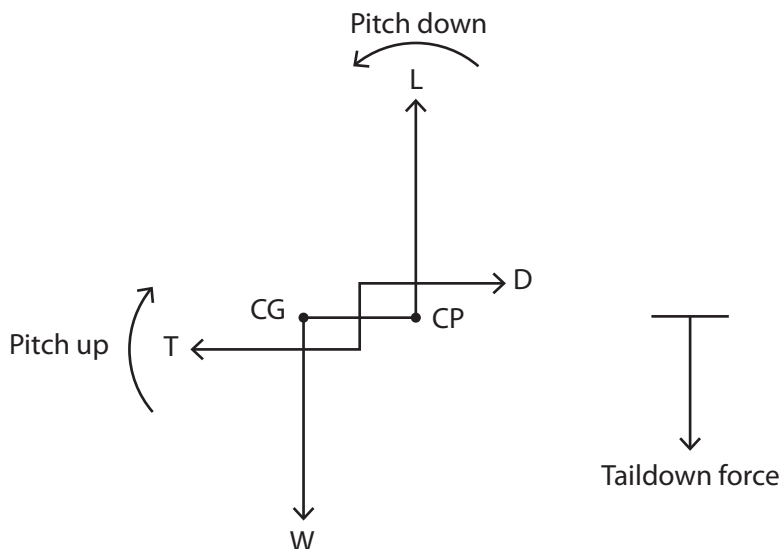


**Diagram 32, Forces in a Descent**

the lift/drag ratio vectors, the angle does not change. Therefore, if the lift/drag ratio remains the same, so too does the glide angle and hence the range on the glide remains the same whatever the weight. Gliding at lower weights can be accomplished at lower speed, but with the glide range, being unchanged this means the endurance on the glide is increased.

**Forces in cruise flight – airplane** Reference to *Diagram 33, Forces in cruise flight – airplane*, shows the four forces present on an aircraft during straight and level flight.

The conventional layout on an aircraft with a low thrust line (under slung engines) coupled with a high dragline is to produce a nose-up moment with power application, shown in *Diagram 33, Forces in cruise flight – airplane*. The lift force acting from the centre of pressure is located aft of the centre of gravity location and it produces a nose-down moment. The two opposing forces produce opposite pitching moments, which almost balance out to



**Diagram 33, Forces in cruise flight – airplane**

maintain equilibrium. Any residual pitch-up or down is balanced by the tailplane force, which is usually a down load.

An aircraft with a high thrust line (high wing turboprops or helicopters) and a low dragline will produce a nose-down couple. The centre of pressure is then required to be located forward of the centre of gravity to produce a nose-up couple. These two couples will then produce equilibrium with the aid of the tailplane's up-force.

See **Tailplane; Diagram 34, Helicopter Equilibrium Cruise Forces.**

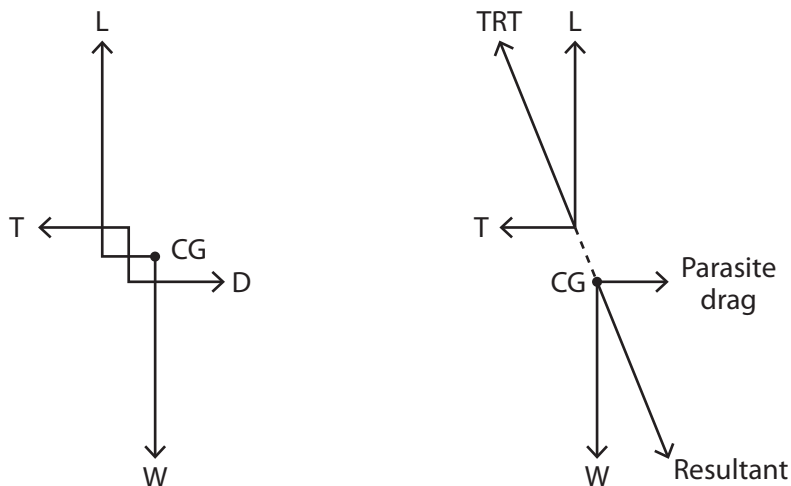
**Forces in cruise flight – helicopter** The forces found in stabilized cruise flight on a helicopter involve the thrust/drag and lift/weight vectors, but their layout is similar to those same forces on a high wing airplane due to the helicopter's high thrust line and low dragline].

The thrust line is higher on a helicopter due to the rotor disc's location and the lower dragline is due to the fuselage. The thrust equals the parasite drag, which increases with speed and the lift (vertical component) equals the weight. This shown by **Diagram 34, Helicopter Equilibrium Cruise Forces.**

During the hover, the lift force is equal and opposite of the weight and both forces are in alignment. With increasing forward speed from the hover, the rotor disc tilts forward, the thrust component increases and parasite drag starts to develop and increase. As the thrust/drag couple becomes established it causes the fuselage to pitch nose-down. This causes the centre of gravity to move slightly aft due to the fuselage tilt. Therefore, the vertical component of lift is forward of the centre of gravity with the weight vector extending below the centre of gravity. The lift and weight vectors are then out of alignment during cruise flight and produce a nose pitch-up moment. Equilibrium is established when the nose-down pitch (thrust /drag) opposes the nose-up pitch due to the lift/weight couple and the total rotor thrust and resultant forces are aligned



through the centre of gravity. Therefore, the centre of gravity's position partly determines the degree of fuselage tilt. The tail stabilizer balances out the residual pitching moments, usually with a down-force as is normally found on an airplane.



**Diagram 34, Helicopter Equilibrium Cruise Forces**

Reference to ***Diagram 35, Rotor Forces in Cruise Flight***, shows the forces present on a helicopter's rotor disc during straight and level cruise flight.

The inflow component and axis of rotation are also included in this diagram. The total rotor thrust (lift) is inclined slightly forward of the axis of rotation. The component acting forward between the lift and the total reaction is the forward thrust – the force that propels the helicopter forward. It is equal and opposite to the helicopters drag. The vector  $V_r$  (RPM) is on the plane of rotation.

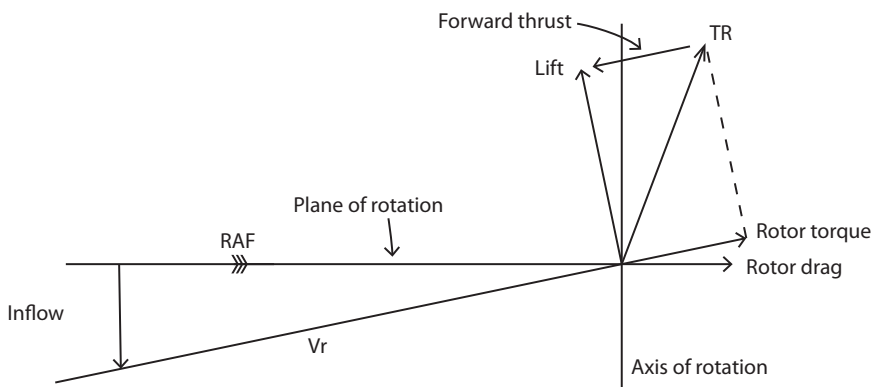
**Forebody strake** A forebody strake is a leading edge root extension of very low aspect ratio mounted along the sides of the fuselage. Their purpose is to produce a strong vortex over the wing to improve handling at high angles of attack, especially on delta type wings or swept back wings. They are commonly found on supersonic fighter aircraft where the strake extends forward below the cockpit canopy.

See **Chine**; **Strakes**; **Vortex lift**.

**Foreflap** In a system of double or triple slotted flaps, the leading flap (closest to the trailing edge of the wing) is known as the foreflap. It has nothing to do with leading edge flaps.

See **Slotted flaps**.

**Foreplane** A foreplane is a wing of short span mounted on the forward fuselage to enhance take-off and landing performance. They may be retractable or fixed, with or without elevators. They are also known as moustaches or more commonly as canard wings.



**Diagram 35, Rotor Forces in Cruise Flight**

**Form drag** Form drag is produced when airflows over a body (aircraft) and the streamlined airflow changes from laminar to turbulent flow resulting in flow separation, form drag and a low-pressure wake flow behind the aircraft. The classic example is a flat plate placed at right angles to the airflow producing turbulent air behind the plate. Streamlining reduces the amount of form drag and the following wake.

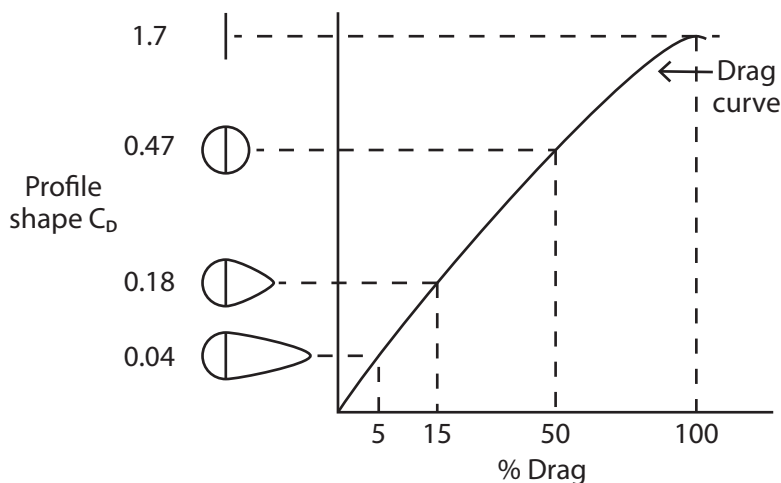
The drag coefficient for various shapes is:

- 1.17 for a flat plate
- 0.47 for a circular tube
- 0.18 for a teardrop shape
- 0.04 for a streamlined shape.

**Diagram 36, Profile Shape V. Percentage Drag**, shows these values. It can be seen the percentage drag reduces with streamlining. Leonardo Da Vinci (1452–1519) noted drag is proportional to the object's area ( $S$ ) and streamlining reduces it.

Also known as boundary layer drag or pressure drag.

See **Streamline shape**.



**Diagram 36, Profile Shape V. Percentage Drag**

**Form thrust** Form thrust is a feature of vortex lift found on delta wing type aircraft. Delta wing aircraft produce a vortex over the wing's upper surface, which contributes to the lift at high angles of attack. The low pressure acts on the wing's leading edge and forward upper surface as a suction or form thrust helping to pull the wing (and plane) forwards. In effect, it acts as a negative drag.

It is also known as lift boost.

See **Vortex lift**.

**Forward centre of gravity limit** The centre of gravity's forward and aft limits is determined by two entirely different factors.

The forward centre of gravity limit is set by the amount of elevator control authority required to rotate the aircraft at low air speeds during take-off and landing. The ability to flare the aircraft for landing normally requires greater stick force than that required for take-off, due to the lack of propeller slipstream over the tailplane and also the increased nose-down moment due to full flap application.

It would be most convenient for aircraft designers, pilots and aircraft operators if the centre of gravity remained close to the middle of the centre of gravity range at all times, but of course, this does not happen. There will be times when it is closer to one limit or the other. There are several advantages, but mostly disadvantages of having the centre of gravity near its fore and aft limits.

The disadvantages of the centre of gravity near its forward limit:

- When the aircraft is about to touch down and is in ground effect, elevator authority is greatly reduced. However, elevator authority must still be powerful enough to pitch the nose up to the required attitude for landing when trimmed to a three-degree approach slope. Pitching the nose up during the landing flare will require a greater force and movement on the control column than it does in free flight and the chance of bouncing on touchdown will be increased. It is a requirement the stick force per 'g' – the

amount you pull back on the stick – and the elevator control forces in relation to trim must always remain within acceptable limits. Too much elevator power permits high angles of attack to be quickly achieved with easy stall entry. The stall entry may be more docile at forward centre of gravity locations compared to an aft centre of gravity, but the actual stall characteristics should remain unchanged. Tail-wheel type aircraft would be more likely to tip on their nose with heavy brake use when taxiing or landing. Finally, trim changes will be larger for any given change in air speed and the large trim tab angles produce unnecessary and unwanted extra drag.

The advantages of forward centre of gravity location:

- The aircraft's empty weight centre of gravity location is preferred to be near the forward limit; the centre of gravity will usually move aft when fuel and payload are added. Increase in the aircraft's static stability.

*Compare with **Aft centre of gravity.***

*See **Diagram 12, Centre of Gravity Range; Centre of gravity (c.g.).***

**Forward slip** A forward slip is the same manoeuvre as a sideslip except for the aircraft's heading, which is first altered to either side of the required track in order to side slip along the original track.

*See **Sideslip.***

**Forward stagnation point** An area of increased pressure forms at the stagnation point on the leading edge of a wing where the boundary layer air is brought to a standstill relative to the wing's leading edge. The remaining air diverts around the stagnation point, over and under the wing. The stagnation point lowers on the leading edge with increasing angle of attack, effectively increasing the wing's upper surface area and therefore, increasing wing lift.

The leading edge suction also increases with increasing angle of attack.

The stagnation point takes its name from the fact the air stands still – stagnates – at the leading edge. Due to the aircraft's forward motion and its frontal shape, static and dynamic air pressure builds up to produce form drag, a hindrance to forward motion. There is also an aft stagnation point.

See *Diagram 65, Pressure Differential & Stagnation Points*.

**Forward sweep wings** Forward sweep is the opposite of sweepback, where the wingtips are further forward than the wing roots.

A wing with either forward sweep or sweepback should in theory produce the same aerodynamic effects. However, although there are several advantages and disadvantages for forward and aft sweep wings, the forward sweep wing has proven over the years to be the less favored of the two choices.

The advantages of forward sweep wings are:

- the air flows inwards towards the roots, not outwards towards the tips
- it is easier to control the stall aerodynamically when the wing stalls near the wing roots first
- at high angles of attack, the tips remain un-stalled
- higher lift/drag ratios and also higher lift coefficients are achieved due to the forward angle of sweep being less than that of a sweptback wing
- a forward sweep wing at low supersonic speed places the wing outside of the Mach wave formed by the wing's root area. The main part of the wing may then still be in a subsonic flow with wave drag reduced compared to one that is swept back
- roll control is greatly improved
- Wingspan and aspect ratio can be reduced saving weight.

The disadvantages of forward sweep wings are:

- forward sweep wings are highly loaded at high cruise speeds, greater than a swept back wing
- no weight saving due to the required strength for resisting wing torsional bending loads
- forward sweep wingtips are more prone to upward bending loads due to dynamic pressure; as the angle of attack at the tip increases the wingtips tend to turn up causing adverse aileron reversal and possible total structural failure of the wing
- unsuitable for winglets
- a larger pitching moment with flaps deployed
- Reduced yaw stability.

As early as 1931, the Langley Memorial Aeronautical Laboratory was researching the forward sweep wing concept. However, it was



**Photo:** The Grumman X-29 research aircraft was built to research the forward sweep wing concept. This aircraft is on display at the National Museum of the United States Air Force in Dayton, Ohio.

not until 16 August 1944, that the German Junkers Ju-287 four-engine, jet bomber became the first aircraft to be built and flown with forward sweep wings; the wings were swept forward at  $25^\circ$ . The first prototype, built in Germany was captured by the Russians at the end of the war and taken back to Russia, where the second prototype was completed and flown there. It was Dr. Hans Focke, head project designer who instigated the use of a forward sweep wing.

The German HFB-32 Hansa Jet was a successful business jet, also designed by Dr. Hans Focke, which made its first flight in 1964. It has a fifteen-degree forward sweep wing, not for stability purposes but to place the main wing spar aft of the main cabin area.

Several sailplanes have been built with forward sweep wings for enhanced pilot visibility. In later years, the Grumman X-29 research/fighter aircraft was built for NASA to further research the forward sweep wing concept, digital fly-by-wire, canards, composite construction, and flight at high angles of attack up to  $67^\circ$  alpha. Two aircraft were built, the first one making its first flight on 14 December 1984. In Russia, Sukhoi built an advanced, fifth generation, experimental jet fighter demonstrator, the Sukhoi SU-47 Golden Eagle. The high aspect ratio, forward sweep wing gives the aircraft a long range, high-speed in excess of Mach 1.6 and a rate of climb at an incredible 46200 feet per minute! It made its first flight on 25 September 1997.

The term swept forward is unacceptable in aeronautical technology as opposed to the term swept back, which is acceptable; use only the term 'forward sweep'.

**See Aeroelasticity; Flutter; Flexural aileron flutter; Mass balance; Reversed controls; Divergence; Divergence speed.**



**Fowler flaps** Fowler flaps are found on all modern jet transport aircraft, plus other aircraft, being the most effective type of flap available. The flaps, on initial application lower to about  $5-15^\circ$  for take-off and move rearwards relative to the wing increasing the chord length and thus the wing area. Further application lowers the flaps to about  $40^\circ$  maximum extension for landing. The first few degrees of application greatly increase the lift with little effect on drag. At full flap, drag is increased by an appreciable amount with only a slight further increase in lift. Fowler flaps produce the greatest increase in lift coefficient compared to other types of flaps and are usually double or triple slotted flaps.

The first production aircraft to use Fowler flaps was the Lockheed 14 Super Electra, built in 1937. Single and double-slotted Fowler flaps are now common on many jet transport aircraft.

*See **Flaps; Area increasing flaps.***



**Photo:** Fowler flaps are the most common types of flap system used on transport aircraft. This photo shows a Boeing 737-800 with its Fowler flaps fully extended.

**Fowler, H.** Harlan Davey Fowler (1895–1982) was an American aeronautical engineer and inventor. He is best remembered for his invention of the area-increasing flaps in 1924 that are now synonymous with his name, the Fowler flaps.

**Fowler movement** The act of the flaps extending rearwards to increase wing area is known as the Fowler movement.

**Free-molecular flow** In contrast to a relatively low-speed continuum flow, the free-molecular flow experiences hypersonic molecular motion where the air molecules are well spaced (rarefied low air density) and collisions between themselves occur less frequently, although they do of course impact the aircraft itself. In a free-molecular flow, shockwaves become relatively thick and indistinct.

Free-molecular flow is found for example, at the outer boundary of the atmosphere and during hypersonic flight of manned and un-manned space vehicles. It is also known as supraerodynamics or Newtonian aerodynamics.

*Compare with* **Continuum flow.**

**Freestanding propeller** A freestanding propeller is one that is theoretically unaffected by the presence of the fuselage or engine nacelle behind the propeller.

In this case, we refer to the propeller thrust as ‘free thrust’. However, if we take into consideration the presence of the fuselage or nacelle and its affect on the propeller slipstream, we then refer to the disturbed slipstream as ‘apparent’ or ‘gross thrust’. Going one-step further, if we subtract the drag from the gross thrust (caused by the slipstream flowing over the fuselage or nacelle), we have ‘propulsive thrust’ or ‘net thrust’. Propulsive thrust is always a constant fraction less than the apparent thrust, due to the drag being proportional to the ‘slipstream velocity squared’.

**Free-stream airflow** An alternate name for **Relative airflow (RAF)**.

**Free-stream air density ( $\rho_{\infty}$ )** The air density about an aircraft unaffected by its presence.

**Free-stream velocity  $V_{\infty}$**  The velocity of the airflow upstream of the aircraft and around the aircraft at some distance and not affected by the passage of the aircraft in any way. The airflow velocity above the wing and around certain parts of the aircraft is greater than the free-stream velocity.

**Free-stream Mach number** The Mach number of the free air stream velocity flowing over the aircraft and unaffected by the aircraft's presence is shown by the following:

$$\text{Free-stream Mach number} = \frac{\text{True Air Speed}}{\text{Local speed of sound}}$$

The free stream Mach number is abbreviated as  $M_{\infty}$ ,  $M_{FS}$ , or simply  $M$ .

It is also known as the flight Mach number.

**Free-stream static pressure** Static pressure, as opposed to dynamic pressure, is the ambient air pressure surrounding an aircraft in flight and unaffected by its presence.

It is denoted by the symbol  $p_{\infty}$ .

**Free thrust** Free thrust is the term used to describe the slipstream-thrust from the propeller when unaffected by a body (fuselage or engine nacelle) behind the propeller.

See **Freestanding propeller**.

**Friction drag** Friction drag, or shear stress, is part of the profile drag and has the same meaning as skin friction drag. Air friction is associated with the airflow in close proximity and flowing parallel to the surface of the aircraft, as opposed to air pressure, which acts at right angles to the surface.

*See Drag; Skin friction drag.*

**Frise, L.** Leslie George Frise (1897–19?) an Englishman, was the chief designer for the Bristol Aircraft Company, Ltd. Many of his aircraft designs are now world famous, such as the Bristol Fighter, Bulldog, Beaufighter and Beaufort, etc. He may be best remembered for being the designer of Frise ailerons in 1920.

**Frise ailerons** Frise ailerons are designed to counteract the effect of adverse aileron yaw during turns.

The Frise ailerons are attached to the wing's trailing edge with inset hinges; the up-going aileron leading edge protrudes into the airflow passing under the wing. Drag is produced causing the aircraft to yaw towards the lowered wing to aid the turn and prevent adverse yaw. The down-going aileron has its upper surface streamlined with the upper wing surface and so therefore, does not produce any appreciable drag compared to its opposite counterpart.

The up-going aileron is normally over balanced but the opposite is true of the down-going aileron, which is under balanced. By connection through the pilot's control system each aileron balances out the other to provide lighter control forces, compared to a non-Frise aileron system. The use of rudder is essential to balance the turn.

Frise ailerons were first used on the Bristol Fighter, Bulldog, Beaufighter, and Beaufort, Boeing B17 Flying Fortress, Bell P-39, and other British and American aircraft of WW II.

*See Adverse aileron yaw.*



**Photo:** Bristol Beaufighters have Frise ailerons. The aircraft shown here is on display at the National Museum of the US Air Force, Dayton, Ohio.

**Frontal area** The frontal area is the physical frontal area (maximum cross section) of the aircraft that is seen when viewed from the front. However, because the aircraft is streamlined, the frontal area as seen by the air stream flowing over the aircraft is somewhat less. The frontal area is used for estimating drag on relatively blunt bodies such as the fuselage. Frontal area is part of the drag formula where:

$$\text{Drag} = C_D \frac{1}{2} \rho V^2 S$$

Where  $C_D$  = drag coefficient

$\frac{1}{2}$  = a constant

$\rho$  = air density  $1.225 \text{ kg/m}^3$  or  $\text{lbs/cu.ft.}$

$V^2$  = aircraft speed in  $\text{m/sec}^2$  or  $\text{FPS}^2$

S = frontal area in square  
meters or square feet.

Wetted or wing area are also used for drag calculations in place of the frontal area for estimating drag of other parts of the aircraft.

See **Flat plate area**; **Equivalent flat plate area**.

**Froude, W.** The British marine engineer William Froude (1810–1879) introduced the idea of the propeller disc that later became known as Froude's Actuator Disc. It considers the air mass, which on passing through the propeller disc experiences a sudden rise in pressure without affecting the increasing slipstream velocity. Along with William John Macquorn Rankine (1820–1872) and A. G. Greenhill, he was responsible for introducing the propeller's axial momentum theory (disc actuator theory) and the propeller's blade element theory when they were working on ship's propeller theories. Note the theory of ship and aircraft propellers is virtually the same because air at subsonic speed behaves very similar to flowing water.

See **Propeller aerodynamics**.

**Froude's efficiency**      See **Ideal efficiency**, **Propulsive efficiency**.

**Full throttle height/altitude**      An aircraft with a normally aspirated engine (non-turbo or non-supercharged) with a constant speed propeller, will reach its full throttle height during the climb, when the throttle is fully forward. Above this height the engine power, as shown on the manifold pressure gauge, will decrease with increasing altitude. A turbo or supercharged engine will continue to provide full power to a much greater height, known as the critical altitude.

**Fully articulated rotor** A fully articulated rotor system found on some helicopters, which allows the rotor blades to lead/lag, flap, and feather through hinges or bearings, allowing full and free movement of each rotor blade individually.

The pitch hinge allows pitch change movement (feathering) as found on a constant-speed propeller. The flapping hinge allows the blade to flap up and down to form a cone and to compensate for dissymmetry of lift. The drag hinge permits the blade to move back and forth (lead-lag) within the plane of rotation of the disc. Lead-lag allows freedom of angular movement of the helicopter's rotor blades to compensate for faulty drag dampers and rotor blades out of balance, which can cause the rotor head mast to oscillate violently.

The fully articulated rotor system was invented by Juan de la Cierva for his autogiro designs. The degree of control power lies



**Photo:** The Aerospatiale AS 350 Ecureuil (Squirrel) has a fully articulated rotor system.

intermediate between that found on a rigid rotor system and a semi-rigid rotor system.

See **Blade flapping**; **Flapping to equality**; *Diagram 45, Inflow (Transverse) Roll*.

**Fully developed flow** A fully developed flow refers to the airflow over the wings. The viscous airflow over the wing is at a constant velocity and the boundary layer thickness is stabilized.

See **Airflow**.

**Fully developed stall** A fully developed stall occurs at the angle of attack where the airflow becomes turbulent and breaks away from the airfoil surface considerably reducing lift. One wing normally drops before the other, giving it the alternate name of wing-drop stall.

**Fuselage** The fuselage is the main load and passenger-carrying compartment of the aircraft. The word fuselage is a French word meaning spindle.

**Fuselage axis** The fuselage axis, being located parallel to the fuselage centreline, is used as a basis for measurements of location of various aircraft components. This is an alternate name for fuselage zero reference plane.

**Fuselage cross flow** At high angles of attack the airflow over the upper fuselage area can roll up into cross flow separation vortices, which then flow over the T-tail increasing the downwash and thus reducing its static longitudinal stability resulting in pitch-up.

See **Pitch-up**.



**Fuselage pitching moments** It is not only the wings and tailplane that produce a nose-up pitching moment; the fuselage itself also provides pitch-up and a degree of lift when the angle of attack is increased. Under the nose of the fuselage, the air pressure in the viscous flow is increased while above the nose the air pressure decreases. The reverse happens at the rear of the fuselage where the air pressure is increased above and decreased below. This causes a destabilizing nose-up pitching moment and some lift to be generated with induced drag. The total pitching moments of the fuselage and wings, etc, determines the size of the horizontal tailplane.

**Fuselage station** A station is a dimension measured in millimeters or inches from the datum to all planes along the fuselage axis. The datum itself is measured as zero and known as the 'station zero'. The fuselage station corresponds to the moment arm.

*See* **Moment arm.**

# G

**G** The capital letter 'G' represents the wing loading stress force applied to an aircraft in flight resulting from acceleration due to manoeuvring. In straight and level flight, the aircraft and its occupants experience an acceleration of 1G, due to gravity. In a 60° angle of bank turn for example, an acceleration of 2G (load factor) will be experienced. Note this is 1G due to gravity plus 1G due to acceleration equaling 2G. Positive G is felt during pull-up manoeuvres, steep turns, or up-gusts and negative G is felt during nose-down manoeuvres or due to down-gusts.

In place of a capital G, a lower case g enclosed in quote marks may be used instead, thus 'g'; having said that, most textbooks on aerodynamics do not include the quote marks.

*See* **Load factor**.

**g** A lower case g (without quote marks) represents the acceleration due to Earth's gravity; it is incorrect to say the force of gravity. The international standard value of  $1g = 9.80665 \text{ m/s}^2$  (32.2 ft/sec/sec) at sea level.

Note, a capital G (or 'g') represents a stress force due to wing loading and a lower case g represents acceleration due to gravity.

*See* **Acceleration due to gravity**.

**Gap** The distance between the top and bottom wings of a biplane measured at the leading edges of the wings is known as the gap.

Also the air space between the trailing edge of a wing and its aileron or flap, etc.

*See* **Gap seals**.

**Gap seals** Gap seals are placed between the trailing edge of the wing and the leading edge of the control surface; e.g., flaps, ailerons, elevators, or rudder. Their purpose is to prevent drag producing air from flowing through the gap.

**Gas dynamics** Gas dynamics is the name given to the branch of aerodynamics, which deals with the effects of very high speed (supersonic/hypersonic) air flow, high temperature, and ionization over the aircraft and through the jet engine.

*See Hypersonic aerodynamics.*

**Gas turbine cycle** *See Brayton cycle.*

**Gates S. B.** Sidney B. Gates (1893–1973) an aerodynamicist at the Royal Aircraft Establishment, Farnborough, UK, introduced the terms neutral point, manoeuvre point, stick force per g, static and manoeuvre margins. He also contributed research on spin manoeuvres and recovery techniques.

**Gauge pressure** This is the pressure shown on a gauge to indicate the difference between pressure in a system and the ambient pressure. Another viewpoint states gauge pressure is equal to absolute pressure minus atmospheric pressure. The decrease in air pressure over a wing's surface and that of the ambient free-stream flow is measured as gauge pressure, unless another reference is stated.

It is also spelt as gage pressure.

**Gay-Lussac's Law** The French chemist and physicist Joseph Louis Gay-Lussac (1778–1850) states in his law, ‘at constant volume, the absolute pressure is directly proportional to its absolute temperature’; this is shown by the equation:

$$P/T = \text{constant.}$$

*See* **General Gas Law; Charles' and Gay-Lussac's Law; Boyle's Law; Ideal gas law; Avogadro's law.**

**g-break** A g-break is a sudden change in direction in the pitching plane, such as a pull-up to a high pitch angle possibly involving an accelerated stall or into a climb, or a nose-drop due to a stall.

*See* **Accelerated stall; High-speed stall.**

**Geared tabs** A geared tab is a type of balanced tab physically attached to the control surface such that the amount of tab movement is related to the control surface movement.

*See* **Trimming devices; Balance tab.**

**General energy equation** The general energy equation is associated with the First Law of Thermodynamics.

*See* **Thermodynamics.**

**General Gas Law** The General Gas Law (also known as the Combined Gas Law) defines the thermodynamic equation of state for gas.

The equation of state involves the physical condition of the gas and combines the three quantities of absolute temperature, absolute pressure, and volume. The equation holds true for most average conditions when the pressure of the gas is not too high or about to condense. Under average conditions, if any one of these quantities varies, the other two quantities will also vary in proportion. Boyle's, Charles' and Gay-Lussac's Laws are written as:

$PV = \text{constant}$  (Boyles Law)

$P/T = \text{constant}$  (Gay-Lussac's Law)

$V/T = \text{constant}$  (Charles' Law).

The three laws combined into one law are known as the General Gas Law for an ideal gas and are written as:

$$P V/T = (k) \text{ constant.}$$

Any gas that follows the General Gas Law is considered to be an ideal gas hence the name Ideal Gas Law. If any constant is removed from the equation (equivalent to holding that quantity constant), the remaining equation is one of the three gas laws. This equation may also be written as  $PV = Rk$ :

Where  $P$  = pressure

$V$  = volume

$T$  = absolute temperature

$k$  = characteristic constant for any one gas.

The General Gas Law can be applied to the workings of a gas turbine (jet) engine's compressor. The compressor provides a large quantity of high-pressure compressed air to the combustion chamber at a reduced velocity.

*See Charles' and Gay-Lussac's Law; Boyle's Law; Gay-Lussac's Law; Ideal gas law; Avogadro's law.*

**General momentum equation** *See Newton's Second Law of Motion, Momentum.*

**General momentum theory** A little used theory on how propellers produce thrust. The more accepted theories are the blade element theory and the axial momentum theory. The vortex theory is redundant.

*See Propeller aerodynamics.*

**Geometric angle of attack** The angle between the free stream relative airflow (RAF) and the airfoil's chord line. It is more commonly referred to as the angle of attack ( $\alpha$ ).

See *Diagram 4, Airfoil Terminology*; **Effective angle of attack**.

**Geometric mean chord** See *standard mean chord*.

**Geometric mean pitch** The geometric pitch is the mean pitch of all propeller blade elements from root to tip.

**Geometric Pitch** The geometric pitch is defined as 'the distance the propeller advances forward in one revolution when the angle of attack of the blades is at zero degrees'.

*Diagram 37, Geometric Pitch* explains this clearly. When the prop is advancing with zero degrees angle of attack, the advance per rev is equal to the geometric pitch. This distance is a definite length measured in inches for a given propeller, which depends on the geometry of the blades, hence the name geometric pitch. To maintain a constant pitch, the angle of each blade section must increase from the blade tip to the blade root in order to obey the law, geometric pitch =  $2\pi R \tan \theta$ . This was explained earlier in the section on Blade Angle. The equation should hold true for the whole length of the prop blade, but if it does not, the geometric pitch is stated for one section of the blade only at the 'standard radius; this point is located at the 75% station along the length of the prop blade from the hub. From the above formula, we can find the geometric pitch of a propeller in inches as follows:

$$\text{Geometric pitch} = 2\pi R \tan \theta$$

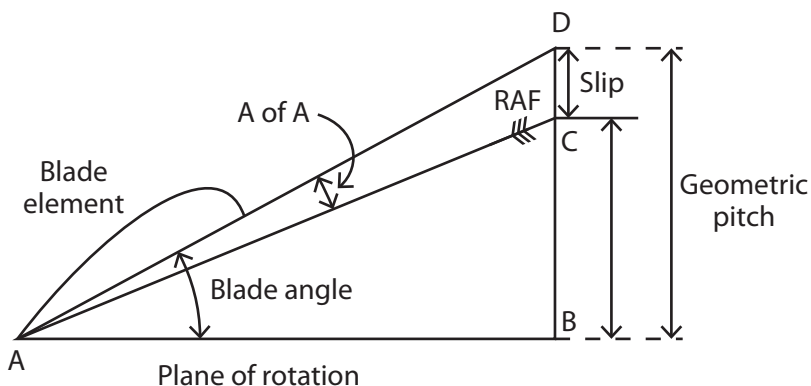
Given:

$$\begin{aligned}\text{Pitch} &= 22^\circ (\tan 0.4040) \\ \text{Propeller radius} &= 26 \text{ inches} \\ \pi &= 3.1416\end{aligned}$$

$$\begin{aligned}\text{Therefore pitch} &= 2 \times 3.1416 \times 26 \times 0.4040 \\ &= 66 \text{ inches.}\end{aligned}$$

An inspection of **Diagram 37, Geometric Pitch**, will reveal a small amount of slip is still present. A small amount of thrust is still generated by the prop at zero degrees angle of attack, which can be attributed to the curved shape of the propeller's back. Generally, the geometric pitch is less than the experimental pitch; however, this may not always be true.

To clarify the points made here regarding the experimental and geometric pitch, the prop blades can be related to the aircraft's wings and tailplane. The vertical tail is a symmetrical airfoil and so both sides are of equal curvature. However, on most aircraft (aerobatic aircraft can be an exception) the main wing is cambered; the upper surface has greater curvature than the lower surface. The propeller blades are shaped similarly. An inspection of **Diagram 49, Lift Coefficient V. Angle of Attack**, shows the symmetrical airfoil section ceases to produce lift at zero degrees angle of attack. However, of greater interest here, at zero degrees angle of attack, the cambered airfoil is still producing lift, or thrust in the case of



**Diagram 37, Geometric Pitch**

the propeller as indicated by a positive lift coefficient. This point corresponds to the prop's geometric pitch. A further reduction in angle of attack to minus two or three degrees will eventually produce zero lift coefficient, known as the angle of zero lift for the wing or the angle of zero thrust for the propeller. This corresponds to the prop's experimental pitch.

*Compare with **Diagram 68, Propeller Pitch** and **Diagram 26, Experimental Pitch** and note the angle of attack and the relative airflow vector. See also **Mean geometric pitch**.*

**Geometric thickness/chord ratio** The geometric thickness/chord ratio is the actual physical measurement of the wing's depth and chord dimensions.

*See **Thickness/chord ratio**; **Aerodynamic thickness/chord ratio**.*

**Geometric twist** The geometric twist is the angle between the propeller blade's chord and its plane of rotation. It varies from blade root to blade tip. It also applies to the wash-in and wash-out of an aircraft's wing.

**Gilruth, Dr. R.** Dr. Robert Rowe Gilruth (1913–2000) joined NACA in 1937 and retired from NASA in 1973. He is credited with being an aviation and space pioneer and 'the Father of Human Space Flight'. He is noted for introducing the term aerodynamic centre and for leading NASA's manned space flight program.

**Glauert factor** The Glauert factor relates the compressibility of the fluid (air) to the increase in lift coefficient.



**Glauert, H.** Despite his German sounding name (his parents were German) the English born Herman Glauert (1892–1934) eventually became the Principle Scientific Officer at the Royal Aircraft Establishment, Farnborough until his sudden death in 1934. He was a Fellow of the Royal Society and an international authority on aeronautical science.

His contribution to aerodynamics was his derivation of the Prandtl-Glauert compressibility correction factor thus:  $(1 - M^2)^{-\frac{1}{2}}$ . The factor is associated with calculations of the effects of compressibility on the lift and drag coefficients over the wings during high subsonic flight. He also researched the problems of flutter and propellers in his many and varied contributions to aerodynamics.

**Glide angle** The glide angle of an aircraft is the angle between the horizontal and the aircraft's flight path.

Flatter glide angles are achieved at greater lift/drag ratios. Lowering the flaps or undercarriage increases the drag reducing the lift/drag ratio and therefore increasing the glide angle. Gliding at any angle other than the optimum will result in less efficiency in glide performance and hence, a steeper glide angle. This is shown by the formula:

$$\text{Glide angle} = \sin \gamma = \frac{T - D}{W}$$

Where  $\sin \gamma$  = angle of descent

T = thrust, lbs or kg

D = drag, lbs or kg

W = weight, lbs or kg.

If thrust (T) is zero, it is cancelled from the formula.

See **Diagram 32, Forces in a Descent.**

**Glide range** The maximum glide range is achieved at the maximum lift/drag ratio shown by:

$$\text{Glide range} = \frac{\text{horizontal distance}}{\text{vertical distance}}$$

Glide range is also known as the gliding distance.

**Glide ratio** An aircraft's glide ratio depends on the lift/drag ratio of the aircraft when the power is zero, i.e., the ratio of the un-powered aircraft's rate of descent to its horizontal distance travelled. If the lift/drag ratio is, for example 15:1, then the aircraft will glide 15 miles forward for every mile of descent.

**Gothic delta** A delta wing in the shape of a Gothic window (as commonly found in churches).

Also known as an ogival delta.

See **Delta wing**.

**Gottingen Aeronautical Research Institute** Gottingen University was the home of Germany's largest institute of aerodynamic research, known as the Gottingen Aeronautical Research Institute. Dr. Dietrich Küchemann, Professor Albert Betz, Professor Ludwig Prandtl, and other famous German aerodynamicists worked there. In 1908, the World's first return airflow, 2-metre-by-2-metre wind tunnel was built and used by Prandtl and others.

**Gouge flaps** The Gouge flaps have a relatively simple flap track system, where the split type of flaps deploy rearwards and downwards on a curved flap track. The wing area is increased along with the wing camber to increase lift and drag. In 1936, Arthur Gouge invented them for use on Short Brother's Sunderland and Empire flying boats.

**Graveyard spiral** The graveyard spiral is an alternate name for spiral divergence.

*See* **Spiral divergence.**

**Gravity axis** The aircraft's OZ gravity axis acts vertically downwards through the centre of gravity. It is also known as the vertical axis.

*See* **Axes; Aerodynamic axes; Vertical axis.**

**Gravitational force** The gravitational force (symbol  $g$ ) is considered to be constant at  $9.80665 \text{ m/s}^2$  ( $32.2 \text{ ft/sec}^2$ ).

*See* **Acceleration due to gravity.**

**Gross thrust** The propeller's slipstream-thrust when disturbed by a body behind the propeller is known as the gross thrust.

*See* **Thrust terminology.**

**Gross wing area** The total area of the wing ignoring the placement of the engine nacelles and fuselage with both wings considered to be meeting at the fuselage centreline is known as the gross wing area.

*See* **Wing area; Reference area.**

**Ground adjustable propeller** A variable pitch propeller, adjustable only when the aircraft is on the ground.

**Ground axis** The aircraft's OZ vertical axis acts vertically downwards through the centre of gravity with yaw being positive when the aircraft yaws to the right around this axis. The OZ ground axis is also known as the OZ normal axis or gravity axis.

*See* **Axes; Aerodynamic axes; Normal axis.**

**Ground cushion** A cushion of air, known as a ground cushion exists when an aircraft or helicopter is flying within a few feet (meters) of the ground. It is formed by an increased mass of air being compressed under the wing ahead of the flaps, which can cause the aircraft to pitch nose down requiring up-elevator to 'hold-off' when landing.

*See Ground effect.*

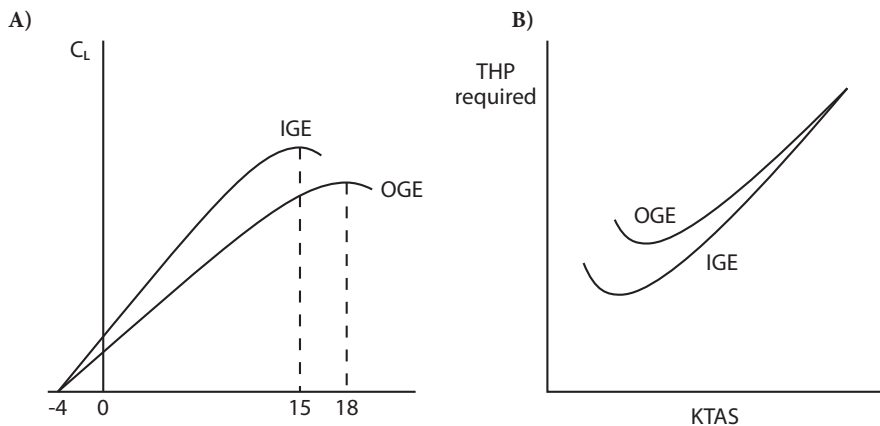
**Ground effect** Both aircraft and helicopters are influenced by ground effect when flying in close proximity to the ground, but by different factors.

When an aircraft is flying close to the ground, particularly when in the flare for landing, the up-wash and downwash angles in the airflow over the wings are both reduced. The lift vector becomes closer to the vertical producing less induced drag relative to an out-of-ground effect situation. This is due to the airflow over and under the wing plus the wingtip vortices flowing parallel to the ground reducing the induced drag and increasing the lift coefficient allowing the aircraft to float further down the runway. When the wing descends below one span height above the ground the reduction in induced drag is increased. **Diagram 38A** shows the lift coefficient versus angle of attack. In ground effect (IGE), the static longitudinal stability and trim is increased aided by the reduced downwash over the horizontal tailplane. A nose-down pitching moment will be induced requiring greater up-elevator deflection to counteract the pitch-down.

When a helicopter hovers within one rotor disc height above ground level, the influence of the ground on the downwash will reduce the inflow velocity and the volume of air as it passes through the rotor disc causing a cushion of air below the helicopter. This produces an increase in the rotor blade's angle of attack due to the reduced inflow component. The total reaction increases and the vector becomes more vertical, moving closer to the axis of rotation because of the reduced inflow angle and the decreasing rotor drag (torque). Therefore, in ground effect, a lower blade angle (collective) and less power is required to hover compared to when out of ground effect (OGE).

**Diagram 38B** shows the thrust horsepower (THP) required versus KTAS when in ground effect is less than the power required when out of ground effect. As the helicopter accelerates to forward flight, the THP required in, or out of ground effect merges to become the same amount.

See **Wingtip vortices**.



**Diagram 38, Ground Effect Changes**

**Ground fine pitch** Some aircraft equipped with propellers that do not have reverse thrust, use an ultra-fine pitch setting to provide extra drag for braking purposes after landing. They are also used on turboprop aircraft to reduce thrust when taxiing and to reduce propeller drag on starting the engine. Aircraft with free-shaft turbine engines do not need ground fine pitch because the propeller is not connected mechanically to the turbine, which is spooled up by the starter motor. A mechanical ground fine pitch stop must be removed by micro-switch activation on the undercarriage, or other means. The act of using ground fine pitch for braking is called 'discing'

See **Discing**.

**Ground resonance** Ground resonance is associated with helicopter operations, where a vibration energy and ground contact is the major requirements for it to develop.

Two vibrations are required for ground resonance to occur. The first is a vibration in the rotor head. On a fully articulated rotor system with three or more blades, which have lead-lag hinges, one or more of the rotor blades move in their drag plane away from the 120° position (assuming a three-blade rotor). Blades of unequal weights and balance, faulty drag dampers or faulty tracking can cause this. The centre of gravity of the rotor disc is then offset from its axis of rotation causing the rotor head mast to wobble. Helicopters with two-blade rotor systems are free of this problem because they do not have, or need lead-lag hinges.

The second vibration of the same frequency is due to the undercarriage being in light contact with the ground, as when landing, or when the helicopter is parked on the ground with the engine idling. The vibrations cause a slow lateral oscillation of the fuselage, which increases in amplitude to a point where the helicopter rolls over, suffers serious damage, and breaks up.

Ground resonance is also known as padding.

*See Dragging; Coleman theory.*

**Groundwash** Groundwash is caused by wake turbulence flowing outwards from the wingtip vortices of heavy aircraft. The propeller wash or thrust from jet engines also causes it.

*See Wake turbulence.*

**g-stall** *See Aerodynamic stall.*

**Guggenheim Aeronautical Laboratory** The Guggenheim Fund for the Promotion of Aeronautics established the Guggenheim Aeronautical Laboratory on 16 June 1926. The Guggenheim Aeronautical Laboratory is located at the California Institute

of Technology as an aeronautical research laboratory. Daniel Guggenheim (1856–1930), a copper tycoon and his son Harry Frank Guggenheim (1890–1971) created the Daniel Guggenheim Fund for the Promotion of Aeronautics with a \$3 million grant.

Theodore von Karman (1881–1963) became its director in 1930 after immigrating to the USA from Aachen, Germany. In 1961, the Guggenheim Aeronautical Laboratory was renamed the Graduate Aeronautical Laboratory.

**Gull wing** Like the sea bird, a gull wing has a sharp dihedral from the wing root extending outboard for a short distance where the wing then extends out to the tips with less dihedral. It is also described as a non-planar wing.



**Photo:** The Martin Mariner has gull wings. This is one of the last examples still in existence and is located at the Pima Air & Space Museum, Tucson, Arizona.

An aircraft that comes to mind with gull wings is the Russian Polikarpov 1-153 Chaika (Seagull) single-engine fighter. Another example is the Martin PBM-5A Mariner twin-engine flying boat, which first flew on 18 February 1939, is shown in the photograph.

*Compare with Inverted gull wing.*

**Gurney flaps** Gurney flaps are typically used on helicopter stabilizers. The Gurney is a very simple, narrow metal strip attached to the rear of an airfoil surface.

The stabilizer's zero lift angle of attack is decreased while the maximum lift coefficient is increased. This effectively increases the airfoil's camber, which therefore increases the nose-down pitching moment. The stabilizer's lift is increased due to a change in the trailing edge Kutta condition. The lift force acts in the downward direction on an inverted stabilizer.

Like any other airfoil, it produces spanwise wake vortices, which consist of pairs of counter-rotating vortices shed alternately in the



**Photo:** A Gurney flap is located on the trailing edge of the inverted horizontal stabilizer of this NATO Helicopter Industries NH 90.



Karman vortex street phenomenon. In addition, at high angles of attack, vortices from the leading edge area may also be produced. The Gurney flap can operate over a wide range of angles of attack making them suitable for helicopter stabilizers. The angles of attack range from about  $+15^\circ$  in autorotation to about negative  $25^\circ$  in a higher power climb.

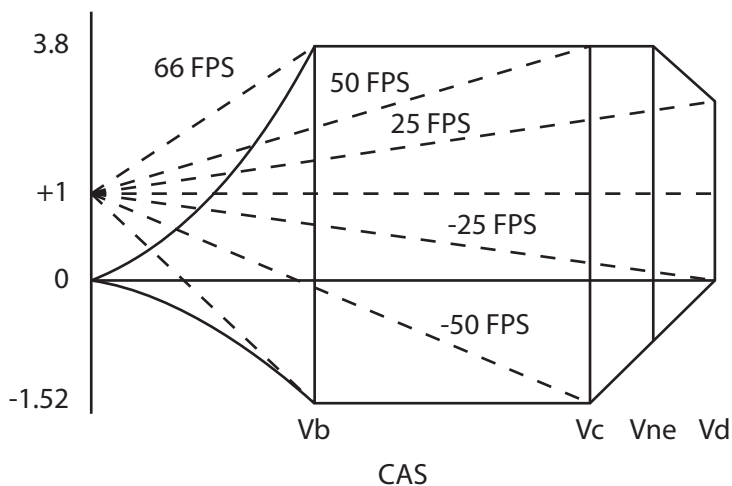
Most, but not all helicopters employ Gurney flaps. They were invented by Dan Gurney for improving racing car performance and are the first aeronautical device to be transferred from the car industry to aviation. Bob Liebeck of Douglas Aircraft named the Gurney flap in honor of Dan Gurney.

**Gust alleviation factor** An up-gust acting on an aircraft in flight will cause the aircraft to rise in the up-draught and the wing will flex upwards. However, the aircraft will continue along at its original flight attitude for several wing chord lengths before the wing generates extra lift due to the increased angle of attack. Therefore, the onset of increased structural load occurs relatively gradually and not instantaneously, as would occur in a theoretical sharp-edged gust, which over-estimates the actual acceleration. The gust alleviation factor takes into account this gradual increase in lift and stress acting on the wing. A sharp-edged gust is one with a high vertical velocity compared to the aircraft's velocity.

**Gust envelope** The gust envelope is plotted as an overlay on a V-n manoeuvre envelope graph representing the gust velocity. The graph shows the operating and limiting speeds are related to wind gusts.

Certification rules define that gusts of 66 feet per second (FPS) are permissible up to the turbulence penetration speed ( $V_B$ ), 50 FPS gusts up to the ( $V_C$ ) cruise speed and 25 FPS gusts up to the design dive speed ( $V_D$ ). The gusts may be in the positive or negative sense.

See **Manoeuvre envelope; Limit & ultimate load factors.**



**Diagram 39, Gust Envelope**

**Gust load factors** A vertical up-gust acting on the aircraft combined with the aircraft's forward speed, will momentarily increase the wing's angle of attack and hence lift. If the lift is increased beyond the load factor design limits, the aircraft will be overstressed. The increase in load factor can be calculated from the following formula:

$$\Delta n = \frac{0.115 m \sqrt{\sigma}}{(W/S) V (SEG)}$$

Where  $\Delta n$  = load factor

0.115 = constant

$m$  = lift coefficient per angle of attack

$\sigma$  = density altitude ratio

$W/S$  = wing loading

$V$  = equivalent air speed

SEG = sharp edged gust in FPS.

**Gyrocopter** See Autogyro

**Gyroscopic effect** The gyroscopic effect of the propeller can cause instability problems, both on take-off and landing. Any spinning mass, the propeller included will be affected by the gyroscopic effect of rigidity. Rigidity is the tendency of the spinning mass (the propeller in this example) to remain with its axis in a fixed position relative to space and to resist any force that tries to move it.

If an applied force succeeds in displacing the propeller's axis, the resulting movement of the mass, known as gyroscopic precession, acts as if the force was applied at a position  $90^\circ$  around the plane of rotation. Relating this last statement to a right-handed propeller on a tail-wheel aircraft, as the tail is raised during the take-off run, it acts as if a force is being applied to the top rear of the propeller causing it to tilt forward. However, because of precession, the propeller acts as if the force had been applied to the rear right-hand side of the propeller disc, causing the aircraft to turn to the left.

The gyroscope force to a greater or lesser degree affects all propeller-driven aircraft when the propeller axis is forced to tilt. The extent of the gyroscopic force will be greater with an increase in propeller weight, RPM, diameter, and the rate of the aircraft's pitch, roll and yaw movements. The effect of this force can be quite noticeable, especially during in-flight manoeuvres. For example, during a steep turn to the left, the plane's nose will tend to rise followed by a climb, although this could be over powered by the aircraft's inherent directional stability leading to a spiral dive. Conversely, a steep turn to the right will cause the nose to drop followed by a descent. It follows; pitching the nose up produces a right yaw, whilst a nose-down pitch will cause a left yaw. Hence, the need to use rudder to maintain balance during certain manoeuvres.

Gyroscopic precession also affects a helicopters rotor system response to cyclic inputs by the pilot.

**See Rigidity & precession of gyroscopes.**

# H

**Haack, Wolfgang Siegfried** Siegfried Haack (1902–1994) was a German mathematician and mechanical engineer. With William Rees Sears (1913–2002) he introduced the Sears-Haack body, a shape determined by mathematics (not ogive) for the use on artillery shells, bullets and the nose shape on jet fighter aircraft.

*See* **Sears-Haack body; Sears, William Rees.**

**Handed propellers** The terms left or right-handed propellers refer to the direction of rotation of tractor propellers when viewed from the rear of the aircraft. For example, a propeller with a blade descending on the left of the disc is termed a left-handed propeller.

*See* **Left-handed propeller; Right-handed propeller.**

**Handley-Page, Sir F.** Sir Frederick Handley-Page (1885–1962) was a British electrical engineer, aircraft designer and manufacturer. He established the Handley Page Aircraft Company in 1909, which remained in business until its demise in 1970. The company built many famous military aircraft such as the HP 0/100 and HP 0/300 (the first strategic bombers used by the Royal Air Force) and followed by the Hampden and Halifax bombers of WW II and later still the HP-80 Victor V-bomber.

The German, Gustav Victor Lachmann (1896–1966) invented the leading edge slot in 1918, to improve high-lift control at low-speed, which was patented on 24 October 1919 and developed by Handley-Page to become known as the Handley-Page leading edge slot. Lachmann also designed the Victor bomber, which Handley-Page named after Dr. Lachmann's middle name of Victor.

*See* **Lachmann, G. V. Dr.; Slots.**



**Photo:** The aircraft shown here, a Handley-Page Halifax at the Yorkshire Air Museum & Allied Air Force Memorial, York, England, is just one example of the many successful designs from Handley-Page Aircraft Company.

**Hargraves, L.** The Englishman, Lawrence Hargraves (1850–1915) discovered in 1884, that the centre of pressure acting on an airfoil lay approximately at the 25% MAC position. He also discovered that the upper surface of the wings should be curved, not flat for better lift and noted the importance of dihedral angle.

See **Centre of pressure**.

**Hayes, W. D.** Wallace D. Hayes (1919–2001) an American aerodynamicist, further developed the area rule concept and also researched the behavior of delta wings operating in the transonic region from 1947 onwards. He also made major contributions to the study of hypersonic flows (Mach 5.0 and above) and sonic booms.

See **Area rule; Delta wing; Sonic boom**.

**Helical flight path** The path followed by a propeller's blade element as it rotates and moves forward in flight will be along a helical flight path. The vector A-C on the right-angle triangle on *Diagram 68, Propeller Pitch*, represents the helical flight path.

See **Helix angle ( $\phi$ )**.

**Helical slipstream** When the propeller rotates, it produces thrust, drag and torque forces. The propeller's drag component causes the slipstream vortex sheet emanating from each propeller blade to be whirled around on a helical path in a 'cork screw' fashion. This leads to a loss in prop efficiency of just under 2%.

Part of the slipstream due to the aircraft's motion through the air, will be flowing straight back over the fuselage. The slipstream generated by the propeller will be rotating. In effect, we can consider the straight slipstream as sliding over the tailplane unnoticed, whereas the helical vortex slipstream will strike the tailplane in a series of pulsations. At a low air speed, the tail will experience more pulsations per unit time than at a high-speed due to the vortex sheets (or corkscrew coils) being close together. This results in the slipstream vortex sheets striking the fin and rudder at a greater angle of attack causing an increase in yaw. At high air speeds, the coils will be elongated and the angle of attack on the tail and fin will be reduced resulting in less yaw.

The rotating slipstream will also strike the underside of the port main wing and horizontal stabilizer at an increased angle of attack increasing the lift. At the same time, the rotating slipstream will strike the upper surface of the starboard wing and horizontal stabilizer at a reduced angle of attack, resulting in less lift. The net result is a rolling moment aiding stability, this time to the right, which under some conditions could counteract the induced yaw to the left, caused by the slipstream striking the fin and rudder.

Also known as the propeller slipstream.

**Helical velocity** The combined speed of the propeller's rotation velocity and the aircraft's forward speed determine the helical velocity of the propeller.

It is also known as the helical tip speed or the race rotation.

**Helical vortex sheet** The helical vortex sheet flowing off each propeller blade affects the following blade by causing a disturbance in the airflow pattern; this results in a loss in propeller efficiency. The greater the number of blades, the greater the disturbance. Using more propeller blades can negate the advantage of greater solidity due to the increased flow disturbance. This is one reason why six blades on a normal propeller are the maximum number that can be used before efficiency deteriorates beyond an acceptable limit. [Propfans and Fenestrons are an exception with a greater number of blades].

See **Helical slipstream**.

**Helicopter** The helicopter is the most numerous of all rotary-winged aircraft.

The inventor Leonardo da Vinci (1452–1519) designed a crude form of rotary winged craft circa 1490, but it was not until 1861 that the first helicopter patent was accepted. However, several attempts were made to design and fly helicopters, some were successful others were not. The Bell Aircraft Company was the first to build and fly a CAA certified helicopter, the Bell 47.

The word 'helicopter' was coined in 1861 by the Frenchman Gustave de Ponton d'Amecourt, based on the Greek *helix* (twisted) and *pteron* meaning wing.



**Photo:** The Bell 47B is an early version of the Bell 47 range looking totally different to the more familiar Bell 47G. This example resides in the Udvar-Hazy Centre National Air & Space Museum, Washington, DC.

**Helicopter aerodynamics** Rotor aerodynamics is an alternate word for helicopter aerodynamics. The subject covers the study of aerodynamics associated to the helicopter; mainly, lift and control by a rotor system above the fuselage.



**Helicopter stability** Helicopters are inherently less stable in flight than airplanes.

Helicopters enjoy stick-fixed static stability but are stick-fixed dynamically unstable in flight. In the stick-free condition, they are statically unstable. During forward flight, the directional stability (yaw) about the OZ normal axis is the most stable of all three axes. Lateral instability (roll) is less than directional stability but greater than pitch stability. Longitudinal stability (pitch) is the most unstable axis on a helicopter. Due to the reduced stability, helicopters are more difficult to fly than fixed-wing airplanes.

See **Static stability; Dynamic stability.**

**Helix angle ( $\phi$ )** The helix angle  $\phi$  (phi), which is also known as the angle of advance, is the angle between the propeller's plane of rotation (A-B) and the resultant direction of the relative airflow (RAF) being the vector A-C in **Diagram 68, Propeller Pitch**.

When the engine is running the propeller will have a rotational velocity dependant on RPM, vector (A-B) on **Diagram 68, Propeller Pitch**, and will travel on a circumference distance equal to  $2\pi R$  in unit time in the plane of rotation. The rotational velocity is also known as the tangential velocity. When the aircraft is moving, forward it will have a forward velocity along the vector B-C the axial component and it will cover a certain forward distance (known as the advance per rev) in unit time depending on the forward speed of the aircraft. Any change in prop RPM or advance per rev will induce a change in the helix angle (AB-AC). The helical flight path followed by the chosen blade element will be along the vector AC or the hypotenuse of the right-angle triangle. The helix angle is related to the effective pitch.

See **Diagram 68, Propeller Pitch.**

**Helix roll angle** The wingtips of a rolling airplane in flight will follow a helical flight path. The helix roll angle is the angle between the wing tip's resultant flight path and the airplane's direction of flight, which determines the helical tip roll velocity.

The values given by the helix roll angle vary from 0.1 to 0.9 depending on the airplane's designed performance requirements. For example a fighter airplane requires a high helix roll angle equal to about 0.09, and a transport airplane's roll performance will be about 0.07.

The tip roll velocity =  $pb/2$  and the airplane's velocity =  $V$ . The helix roll angle is therefore defined as:

$$\text{Helix roll angle} = pb/2V = (\text{radians})$$

Where  $p$  = rate of roll in radians per sec  
(1 rad = 57.3 degrees)

$b$  = wing span in feet

$V$  = velocity in ft. per sec.

**Helmholtz, H.** Hermann von Helmholtz (1821–1894) was a German physicist, physiologist, mathematician and a leading scientist of that era.

He is credited with many inventions and discoveries, in particular, the study of vortex motion in fluid flows, which became known as the Helmholtz vortex theorem.

In conjunction with Frenchman Charles Renard, they discovered it was impossible for human powered flight; wings operated by arm power would not support a human.

**Helmholtz vortex theorem** Herman von Helmholtz (1821–1894) published his vortex theorems in 1858, which was later confirmed in 1867 by Lord Kelvin (1824–1907). The theorems deal with the basic behaviour of vortices in three-dimensional, incompressible, inviscid flow. The vortex theorems state:

- along the length of a vortex, the strength remains constant due to its circulation (as in wingtip vortices)
- a vortex filament must extend to the limits of the fluid to form a closed circuit
- The vortex is self-contained; it does not drag surrounding air into itself.

Vorticity and circulation are closely related but are two entirely different phenomena.

See **Bound vortex**; **Kutta-Joukowski theorem**; **Horseshoe vortex**; **Lifting line theory**; **Starting vortex**; **Wingtip vortices**.

**High-energy rotor blades** Helicopter rotor blades may have a mass weight at the blade tips to increase their centrifugal force and energy, which in turn reduces the rotor disc coning angle due to the increase in blade inertia. Rotor RPM remains more constant compared to non-weighted tips, which are an advantage especially during the landing flare. The disadvantage of having the mass weights and the associated greater centrifugal force is thus; a heavier rotor head assembly is required, which makes it impractical for most lightweight helicopters.

**High hypersonic speeds** The high hypersonic speed range is from Mach 10.0 to Mach 25.0 as defined by NASA.

**High bypass engine windmilling drag** The windmilling drag of a high bypass ratio gas turbine (jet engine) can be calculated from the following:

$$\text{Drag}_{\text{windmilling}} = 0.0044 p A_c$$

Where  $p$  = ambient static pressure

$A_c$  = engine inlet area.

**High-incidence stall** With all high-lift devices extended (flaps, slats, etc) a high-incidence stall is entered as the aircraft's speed is progressively reduced until it stalls at its critical angle of attack.

*See all references to stalling.*

**High lift** High lift is achieved by use of high-lift devices to increase the lift to a higher value than that used in the normal cruise configuration, such as during take-off and landing.

*See High lift devices; Configuration.*

**High lift devices** Wings designed for high-speed aircraft may not produce sufficient lift at the low speed required for take-off and landing. To overcome this problem, high lift devices are required such as flaps, leading edge slat, slots, Krueger flaps, etc, to increase the wing area and/or increase the maximum lift coefficient and wing camber. High lift devices also reduce the angle of attack and stall speed for a given lift coefficient, which in turn lowers the take-off and landing speeds. The required lift can also be achieved at the same angle of attack, with less dynamic pressure when the maximum lift coefficient is increased.

Knowing the basic stall speed for flaps up, the flaps down stall speed can be found from the following formula:

$$V_{sf} = V_s \sqrt{C_{Lm}/C_{Lmf}}$$

Where  $V_{sf}$  = stall speed flaps down

$V_s$  = stall speed flaps up

$$C_{Lm} = C_{Lmax} \text{ flaps up}$$

$$C_{Lmf} = C_{Lmax} \text{ flaps down.}$$

**High-speed aerodynamics** High-speed aerodynamics involves all speed above a figure of approximately 200 Knots (Mach 0.3). Above this speed, compressibility must be taken into consideration. As the speed increases, transonic, supersonic and then hypersonic conditions are encountered with their individual peculiarities, which are covered elsewhere in this book.

*See* **Transonic airflow; Supersonic airflow.**

**High-speed buffet** Aircraft designed for subsonic flight that exceed their critical Mach number can experience high-speed buffeting. Shockwaves forming on the wing cause separated, turbulent flow, which strikes the elevator causing a vibration, or buzz followed by a high-speed buffet and/or reduced possible loss of effectiveness.

Also known as the high Mach bucket.

*See* **Buffet boundary.**

**High-speed stall** A high-speed stall occurs at a speed higher than the normal basic flaps up stall speed ( $V_{S1}$ ). This may happen during a steep turn, a sudden pull-up from straight and level flight, or any abrupt flight manoeuvre.

*See* **Accelerated stall; g-break.**

**High subsonic speeds** An airplane is considered to be flying in the high subsonic speed region when the local airflow over the wing reaches Mach 1.0. At this speed, trim changes (Mach tuck), handling and compressibility problems may occur. Supercritical wings enable modern jets to cruise at higher subsonic speeds (Mach 0.9 or more) than the earlier first generation jets whose limit was about Mach 0.8.

*Compare with* **High-speed aerodynamics.**

**High wing** A high wing aircraft is one with the main wing mounted above the fuselage.



**Photo:** The Fairchild C-123K Provider is a US military transport of the Viet Nam era. The high wing and low-set fuselage made loading and unloading convenient. It was powered by two piston-engines plus two turbojet engine.

**Hinge moment** The hinge moment ( $C_h$ ) of a control surface is determined by the distance of the centre of pressure from the hinge line. It determines the ease or heaviness to move the control surface, known as the stick force per g, which is proportional to the true air speed squared.

**Hingeless rotor** This is an alternate name for a rigid rotor system as found on some helicopters.

*See Rigid rotor.*

**Hoerner Dr. S. F.** Dr. Sighard F. Hoerner was a German aviation pioneer, aircraft designer and an aerodynamicist for the Fiesler Corporation where he designed the Fieseler FI 156 Storch (a single-engine STOL aircraft) in WW II. After the war, he moved to the USA and worked for the Douglas Aircraft Company. He was one of the first aerodynamicists to note the presence of wing tip vortices and their adverse effect on aircraft performance; for this reason, he invented the drooped wingtips known as Hoerner tips and later wrote and published a paper on them in 1952.

See **Hoerner tips; Wingtip vortices.**



**Photo:** The German, Fieseler FI 156 Storch was one of Dr. Hoerner's most famous aircraft designs. This example is on display at the Royal Air Force Museum, Cosford, England.

**Hoerner tips** Wingtips with a curved downward droop, were introduced by Dr. Sighard F. Hoerner for use on sailplanes. The droop altered the airflow around the wingtips and directs the vortices away from the ailerons and tips. The tips are rounded on the lower surface with a sharp corner on the upper surface, which prevents the secondary airflow from rounding the sharp tip and pushes the wing tip vortex further outboard, in effect increasing the wing's aspect ratio. This had the advantage of increasing the aircraft's rate of climb, cruise speed, range and lateral stability. His idea was carried a stage further by Richard Whitcomb (1921–2009) when he introduced the now common winglets.

Hoerner type tips are used on some single-engine Cessna light aircraft.

See **Wingtip devices; Winglets.**

**Hooke's universal joint** Robert Hooke (1635–1703) was an English scientist who invented, amongst many other achievements, the Hooke's universal joint.

The Hooke's universal joint is a constant-speed joint, not a constant velocity joint. For this reason, it is no accident it found its way onto helicopters. The Hooke's joint is located between the engine's drive shaft and the rotor hub, which must be free to tilt in various directions. The joint allows the axis of rotation to operate even if it is not in line with the shaft axis. The reason why helicopter blades move on their drag hinges is due to the effect of the Hooke's joint.

The blades move forward and backward on their drag hinges as they accelerate and decelerate respectively around the tip path plane, which tilts in various directions to provide manoeuvrability while the drive shaft rotates at a constant RPM. To allow for the blade's changes in rotational speed, the blade will flap upwards as it moves forward on its drag hinge and flap downwards when it moves rearwards, relative to the direction of disc tilt. The flapping up and down conserves the angular momentum.



**Horizontal stabilizer** The horizontal stabilizer is always mounted at the rear of the fuselage aft of the main wings. Its purpose is to provide trim balance for the main wings as and when required and also to house the vertical fin/rudder and elevator. Mounted at the rear, the stabilizer more often than not, produces a download.

A canard mounted on the forward fuselage, is not technically a stabilizer in the correct sense of the term, although it does assist the aircraft's longitudinal stability. The canard is designed to always provide an uplift to assist the main wings to lift the aircraft and to stall prior to the main wings reaching a stall.

It is also known as the tailplane in USA terminology.

See **Canard wings; Tail load; Tailplane.**

**Horn balance** A horn balance is a short span section of a primary control surface (usually the rudder or elevator) extending forward of its hinge line.

Its purpose is to relieve some of the stick force required by the pilot to move the control surface. When the control surface is deflected, the horn balance area being forward of the hinge line moves into the air stream (assume it moves upwards) where the airflow strikes it and helps to move it further upwards. The part of the control surface behind the hinge line will move downwards in unison as required by the pilot and vice versa. It works in the same manner as an inset hinge where the centre of pressure is located closer to the hinge line, reducing the stick force.

The horn balance may be the shielded, partially shielded or the non-shielded type. The shielded type has a part of the primary control surface (the horn balance) located ahead of the hinge line but shielded by the fixed portion of the tailplane or the tailfin ahead of it.

The non-shielded type of horn balance has the outer portion of the control surface (the horn) lying alongside the tip of the fixed tailplane's main surface. Some aircraft have partially shielded horn



**Photo:** Non-shielded horn balances mounted on the elevators of a DeHavilland DH.114A Heron. Note also the trim tabs.

balance as found on early versions of the Supermarine Spitfire's elevators; later Spitfire versions had un-shielded elevator horn balance as also was the horn balance on the rudder of all Spitfire versions.

If a mass balance is also required on the primary control surface it can be mounted within the horn balance area. A horn balance used on a light aircraft's elevators or rudder has little effect on profile drag when deflected.

One of the first airplane's to benefit from the horn balance was the Bleriot XI monoplane in 1910. A few years later in 1918, the Curtiss F-5L flying boat also used rudder and aileron horn balances.

*Compare with* **Inset hinge; Balance tab.**

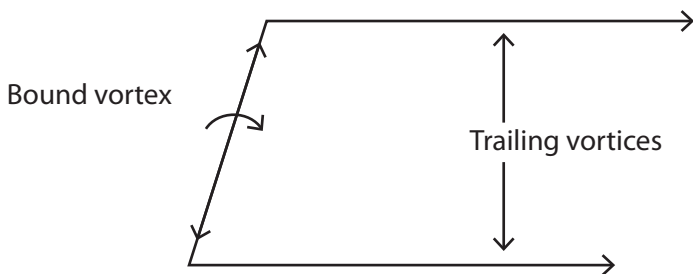
**Horsepower** James Watt the Scottish mining engineer introduced the term of horsepower, where:

1 horsepower = 550 feet-pound per second = 33 000 feet-pound per minute.

**Horseshoe vortex** The horseshoe vortex is a simplified circulation model assuming constant vorticity. The circulation consists of a bound vortex sheet flowing off the wing as a trailing edge vortex coupled with wingtip vortices.

The circulation diagram is considered to be U-shaped with square corners, flowing outward from the aircraft's centreline to the wingtips as a constant circulation bound vortex, with wingtip vortices emanating from the wingtips. This is the horseshoe vortex model as proposed by the Kutta-Joukowski theorem. The bound vortex strength is considered to be zero at the aircraft's centreline and increases in strength to a maximum value at the wing tips.

See **Bound vortex; Helmholtz vortex theorem; Kutta-Joukowski theorem; Lifting line theory; Starting vortex; Wingtip vortices.**



**Diagram 40, Horseshoe Vortex**

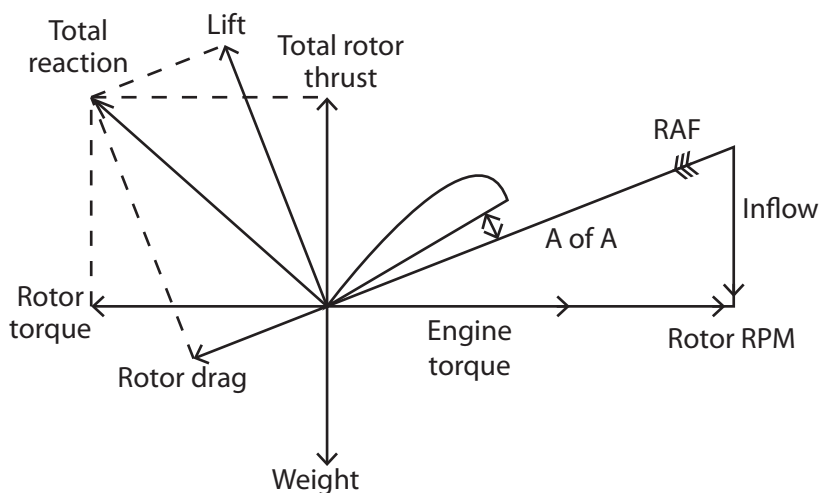
**Hotel** The term hotel is a reference to a pilot's use of an arrested propeller system to stop the propeller rotation while its free-shaft turbine engine continues to run.

**Hovering** Hovering is the ability of a helicopter to remain airborne and more or less stationary, over a fixed location on the ground and at a constant height.

The helicopter has directional static stability in the hover but no dynamic stability. Stability, both static and dynamic, increase with increasing air speed. The vertical component of the total rotor thrust is equal and opposite to the weight; an increase in weight requires an increase in total rotor thrust.

The hover is said to be in ground effect (IGE) when hovering above the ground at a height equal to, or less than the diameter of the rotor disc; the rotor disc downwash is influenced by the presence of the ground. Above this height, the hover is out of ground effect (OGE) where the downwash is free from the reaction from the ground.

See **In ground effect (IGE); Out of ground effect (OGE).**



**Diagram 41, Hovering Forces**

**Hunting** The continual oscillating tendency of a helicopter's rotor blades to lead or lag around a constant neutral point due to the Coriolis force, is known as hunting. An airplane may also hunt around the pitch and yaw axis; roll axis is rarely affected.

**Hypersonic aerodynamics** High temperature, increased drag, chemical dissociation and ionization characterize hypersonic flow. The airflow condition at hypersonic speeds is of importance to re-entry vehicles and rockets. Hypersonic research is known as aerothermodynamics, or gas dynamics.

Hypersonic aerodynamics covers the speed range from Mach 5.0 (3000 MPH) up to Mach 25.0 where the temperature rises in excess of 1000°C (1800°F) due to surface friction heating on the aircraft or re-entry vehicle. Although it should be noted, at speeds as low as Mach 3.0 hypersonic characteristics can make their presence felt. The change in characteristics is gradual from Mach 3.0 through to Mach 12.0, much like the change is gradual through the transonic speed range. Very thick surface boundary layers and high surface temperatures are common to hypersonic flight. At such high speeds, the increased temperature through the shockwave and over the whole aircraft must be taken into consideration by the aircraft designer.

The chemical compositions of the diatomic gas molecules – nitrogen and oxygen – dissociate (break up) in the rarefied air at speeds above Mach 7.0. and the heat destroys the molecular bond between the elements causing separation and ionization to occur. Convective, radiative heat flux and electrically charged plasma forms around the aircraft above a speed of about Mach 12.0 . The heat also causes a large rarefaction in the air density and pressure surrounding the aircraft. Extremely high temperatures at the nose of the craft can be dissipated (viscous dissipation) to some extent by using a blunt nose vehicle (as found for example, on the Space Shuttle) to form a detached shockwave. The temperature is proportional to the square of the Mach number. Heat resistant materials are required to withstand the extreme temperatures that

prevail under these conditions. The following table shows the chemical change of the airflow with increasing temperature:

| Temperature (K) | Chemical change                                                                      |
|-----------------|--------------------------------------------------------------------------------------|
| 800             | molecular vibration                                                                  |
| 2000            | oxygen molecules ( $O_2$ ) dissociate                                                |
| 4000            | formation of nitric oxide (NO)<br>and the nitrogen molecules ( $N_2$ )<br>dissociate |
| 9000            | oxygen and nitrogen atoms ionize.                                                    |

The dynamic air pressure at hypersonic speeds can be as high as twenty five times that of subsonic speeds. Such high air pressure can sustain flight in the very low-density air; although of course, it does require highly sweptback, wedge-shaped wings, and a blended fuselage/wing is an asset at such high speeds. Induced shock waves and expansion waves cause the variations in air density and pressure. At hypersonic speeds at  $4^\circ$  angle of attack the lift/drag ratio reduces down to about 5.5 compared to about 12–15 for subsonic speeds. A decrease in overall stability must also be taken into account due to shockwaves weakening the aerodynamic forces.

It was during 1952–3 and through to the 1970s the US Military services and NACA jointly began studies into hypersonic flight research. The research program involved the development of the North American Aviation X-15 research aircraft in 1961, which was the first airplane to reach and exceed Mach 5 and enter the world of hypersonic flight, piloted by Air Force Major Robert White. The X-15 was purpose-built to explore the problems associated with re-entry from space at high Mach numbers. During the research flights, the X-15 achieved an altitude of 354,200 feet (67 miles) on the fringe of space and a maximum speed of Mach 6.7 (4520 MPH). Prior to this flight Russia's Cosmonaut Major Yuri Gagarin was the first person to achieve hypersonic speed in April

1961 when he made the world's first orbital space flight. This was followed by American Astronaut Alan Shepard's sub-orbital flight a month later. In more recent years, NASA's unmanned X-43A research aircraft achieved a world record speed of Mach 9.6 (nearly 7000 MPH) on 16 November 2004. ICBM's, rockets and the Space Shuttle also operate in the hypersonic speed range.

The term hypersonic was introduced during the 1970s.

**Hypersonic curves** The values of lift coefficient, drag coefficient and the lift/drag ratio in hypersonic flow all take on different characteristics to their subsonic counterparts, as listed below:

- during hypersonic flight, the maximum lift coefficient is achieved at about 55 degrees angle of attack, compared to about 15 degrees for a subsonic airfoil

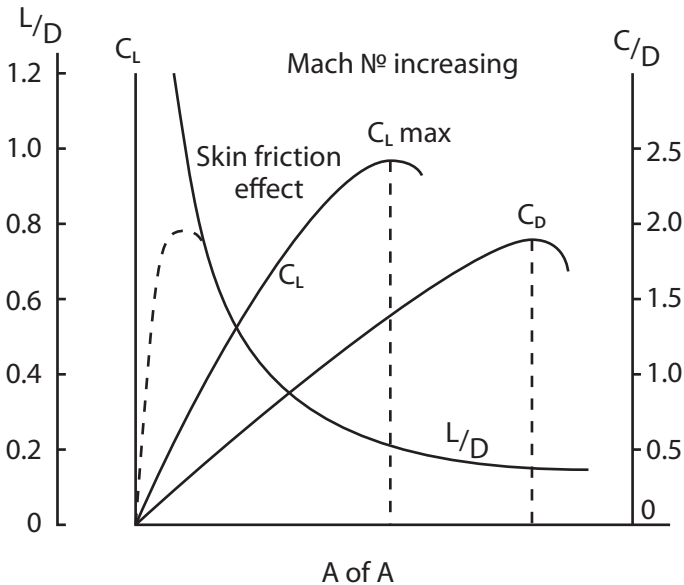


Diagram 42, Hypersonic  $C_L$ ,  $C_D$  &  $L/D$  Ratio

- unlike the subsonic airfoil, the lift coefficient curve for the hypersonic airfoil is initially curved (nonlinear) at low angle of attack
- the drag coefficient curve also differs to the subsonic curve. The hypersonic curve shows a low drag coefficient at low angles of attack and increasing linearly up to a drag coefficient of about 2.0 at 90 degrees angle of attack
- the hypersonic lift/drag ratio reaches its maximum value at low angles of attack and decreases with an increase in angle of attack. The curve is modified by the influence of skin friction at low angles of attack (shown by the dotted line).

*Compare with* **Lift/drag ratio; Lift coefficient & angle of attack; Drag coefficient.**

**Hypersonics** Hypersonics is the study of the flight regime above Mach 5.0 to Mach 25. The term was introduced to aerodynamics in the 1970s and it is classed as a subset of high-speed supersonic flight. The characteristics of hypersonic flow are:

- **Aerodynamic heating**
- **Entropy layer**
- **Low air density effects**
- **Real gas effects**
- **Shock layer.**

*See individual entries.*



# I

**IAS/TAS ratio** When flying at a constant power setting during the climb with a constant Indicated Air Speed (IAS), the True Air Speed (TAS) increases at the rate of 1.6% or approximately 1 Knot per 1000 feet of altitude climbed. The IAS is proportional to the square root of the air density. At 40000 feet, the TAS will be approximately double the IAS, allowing highflying aircraft to fly faster without an increase in drag.

**Ideal efficiency** Propeller efficiency is at its greatest value when the propeller slipstream velocity is close to the aircraft's True Air Speed, the value of the inflow factor is equal to one. At other speeds, where the slipstream velocity is less than the aircraft's velocity (which is the usual case) the value of the inflow factor will of course, be less than one. This is the ideal efficiency of the propeller, where the thrust is related to the airplane's speed and the inflow factor.

It is also known as Froude's efficiency.

**Ideal gas law** In the ideal gas law, the gas pressure varies linearly with volume, temperature, and pressure. The General Gas law is given by the combination of Boyle, Charles', and Gay-Lussac's laws. With the addition of Avogadro's law (the inclusion of the molecular weight of the gas) the ideal gas law is formed, given as:

$$PV = nRT$$

Where: P = pressure of gas

V = volume of gas

n = molecular weight of gas

R = universal gas constant

T = temperature (absolute).

The Frenchman Emile Clapeyron (1799–1864) combined the Boyle's, Charles', and Gay-Lussac's laws in 1834 to produce the ideal gas law.

See **General Gas Law; Charles' and Gay-Lussac's Law; Boyle's Law; Avogadro's law.**

**Ideal pitch** At a negative angle of attack, the propeller produces zero net thrust at the experimental pitch. From the airplane designer's point of view, it is the ideal propeller pitch. On **Diagram 68, Propeller Pitch**, it is shown as angle (AB-AE).

Also known as the exponential mean pitch.

See **Experimental pitch.**

**Impact pressure** In a compressible fluid flow at speeds above 200 KTAS/Mach 0.3, the impact pressure is defined as the difference between total pressure and static pressure. The impact pressure is denoted as  $q_c$  or  $Q_c$ . It is the equivalent of dynamic pressure, which is used in an incompressible flow at speeds below 200 Knots/Mach 0.3.

It is also known as dynamic pressure or compressible dynamic pressure.

**Impact temperature** When the aircraft's speed increases towards the supersonic range the increased air pressure acting on forward facing parts of the aircraft causes a rise in temperature, known as the impact temperature.

See **Aerodynamic heating; Kinetic heating; Temperature rise.**

**Incidence** The incidence of the wing is the angle between a wing's chord line and the airplane's OX longitudinal axis. Since the wing itself flies at a given angle of attack, the fuselage takes up the pitch attitude depending on the incidence (angle). As a passenger

in a jet transport, you may have noticed the fuselage is angled up-hill towards the front of the plane. Incidence, or angle of incidence, is sometimes mistakenly used for angle of attack; angle of incidence is constant, and angle of attack is variable. It is also known as the wing incidence.

*See* **Angle of incidence.**

**Incipient shockwave** An incipient shockwave forms on the upper surface of a wing in subsonic flow when the local velocity reaches the critical Mach number for that airfoil. A subsonic flow is commonly found behind a normal shockwave. The shockwave lies at right angles, or normal, to the surface and is stronger on a conventional airfoil than on a supercritical airfoil. Supercritical airfoils are rear-loaded and produce a shockwave further aft on the wing.

The terms incipient shockwave or weak shockwave are alternate names for a normal shockwave.

*See* **Normal shockwave.**

**Incipient spin** An incipient spin develops when the aircraft wing stalls with the aircraft in a yaw condition. The spin can be prevented at this stage with correct and timely recovery technique.

*See* **Stall-spin.**

**Incipient stall** An incipient stall is the first stage of stall development, characterised by decreasing airspeed, sloppy controls, stall warning activation, and vibration felt through the control yoke as the turbulent flow off the main wings strikes the tailplane.

**Inclined shockwave** At speeds slightly higher than Mach 1.0, shockwaves are initially vertical but become inclined shockwaves as the Mach number increases.

*See* **Oblique shockwave; Diagram 15, Compression Flow & Oblique Shockwave.**

**Incompressible flow** The air is assumed an incompressible, or constant density airflow at all speeds below 200 KTAS/Mach 0.3. The incompressible airflow acts like flowing water; the effects of air density are negligible due to only small changes in temperature and pressure. Air speed is the variable; an increase in airflow speed above 200 KTAS/Mach 0.3 increases the local air density and becomes compressible.

In a streamlined incompressible flow, four types of energy are present in the form of heat, pressure, kinetic and potential energy. The continuity and momentum equations are relevant to incompressible flow.

*Compare with* **Compression flow.**

**Ideal gas law** *This is also known as the* **General Gas Law.**

**Index units** The reduction factor is a constant, which is divided into a moment to produce an index number. By using powers of ten (10, 100, 1000, 10000 etc) large moment numbers can be reduced to a more manageable smaller number. It is also known as the reduction factor.

**Induced angle of attack ( $\alpha_i$ )** The basic angle of attack is defined as the angle between the free stream relative airflow (RAF) and the chord line, comprising the induced angle of attack ( $\alpha_i$ ) plus the section angle of attack ( $\alpha_o$ ); therefore the angle of attack,  $\alpha = \alpha_i + \alpha_o$ .

The section angle of attack ( $\alpha_o$ ) is the difference between the average relative wind and the airfoil's chord line. The induced angle of attack ( $\alpha_i$ ) is defined as the difference between the free stream relative airflow (RAF) and the average relative wind, which is the direction of the modified airflow due to its deflected passage over and under the wing. The section and induced angles of attack each make up half of the basic (or geometric) angle of attack ( $\alpha$ ). The induced angle of attack ( $\alpha_i$ ) is dependant on the lift coefficient

and the aspect ratio; the angle of attack reduces with a decrease in the aspect ratio. The induced angle of attack can be found from the following formula:

$$\text{Induced angle of attack } (\alpha_i) \text{ in degrees} = \frac{18.24 C_L}{AR}$$

Where  $C_L$  = lift coefficient

AR = aspect ratio

18.24 = constant.

See **Induced drag; Diagram 43, Induced Drag; Elliptical lift distribution.**

**Induced downwash** The airflow passing over and under the wing is induced to flow downwards off the trailing edge; hence, the term induced downwash. The amount of downwash is determined by the wing's lift coefficient and its aspect ratio.

See **Downwash; Wingtip vortices.**

**Induced drag** Induced drag is lift dependent drag, due to the difference in air pressure above and below the wing. It depends primarily on the lift per unit span and varies as  $1/\text{speed}^2$ . Whenever lift is being produced, induced drag will always be present. It is caused by the induced downwash over the wings, producing a vortex sheet and wingtip vortices, which in turn increase the effective angle of attack and downwash.

The remote flow direction is modified by being deflected further downwards, increasing the downwash angle. Therefore, the total reaction is tilted further back to maintain its position at right angles to the average relative wind direction over the wing. The lift vector's horizontal component represents the induced drag, also known as the trailing vortex drag.

The wing planform and hence the aspect ratio, has a greater influence on the induced drag than wing taper or area alone. A

greater aspect ratio wing has relatively less area affected by wingtip vortices and therefore, less induced drag. An elliptical or tapered wing produces the least induced drag. Conversely, straight wings produce about 10% more induced drag due to a less than perfect elliptical load distribution. Winglets and washout also help to reduce induced drag.

Increased weight requires an increase in lift, resulting in greater induced drag, which varies with the square of the lift produced. An increase in altitude also increases induced drag. At greater altitudes, the air density decreases, reducing lift; this requires a greater angle of attack to maintain lift and hence, greater induced drag is produced. This is the reason for winglets on high-flying jet aircraft.

Induced drag can be reduced by incorporating reduced wing camber, reduced chord length, wingtip wash-out, taper and increased aspect ratio, as mentioned above.

From the above, it can be seen the induced drag:

- is lift dependent drag
- varies as the square of the lift coefficient
- wing planform and aspect ratio has a greater influence
- it varies inversely as the square of the speed, increasing with speed reduction
- it is inversely proportional to the aspect ratio
- Increases with angle of attack.

Induced drag can be calculated from the following formula:

$$\text{Total induced drag (D}_i\text{)} = (C_L^2 / \pi \text{ AR}) \frac{1}{2} \rho V^2 S$$

Where:  $C_L^2$  = lift coefficient squared

$\pi$  = 3.14 constant

AR = aspect ratio

$\frac{1}{2}$  = a constant

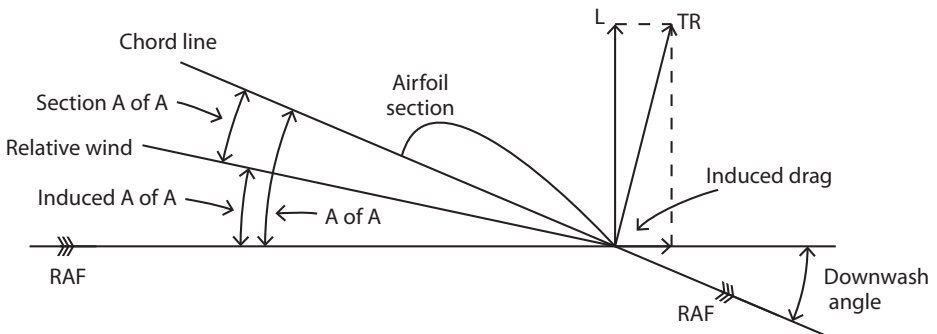
$\rho$  = air density kg/m<sup>3</sup> or lbs/cu.ft.

$V^2$  = aircraft speed in m/sec<sup>2</sup> or FPS<sup>2</sup>

S = gross wing area in square  
meters or square feet.

Prandtl gave the term induced drag circa 1917, to describe the drag caused by the generation of lift. The aeronautical term 'induced' is taken from the term induced as used to describe electrical currents induced by electro-magnetic fields. Induced drag is also known as the trailing vortex drag.

See 'K'; *Diagram 6, Aspect Ratio & Vortex Drag; Diagram 20, Total Drag Curves.*



**Diagram 43, Induced Drag**

**Induced drag coefficient** The induced drag coefficient represents the ratio of induced drag to dynamic pressure multiplied by the wing area. It is proportional to the lift coefficient squared ( $C_L^2$ ) and inversely proportional to the aspect ratio. For planforms other than an elliptical shape, the formula includes the span efficiency factor and is written as:

$$\text{Induced drag coefficient } (C_{Di}) = \frac{C_L^2}{\pi AR e}$$

Where  $C_L^2$  = lift coefficient squared

$\pi$  = 3.14

AR = aspect ratio

e = span efficiency factor (0.7 to 1.0).

The value for the span efficiency factor 'e' varies from 0.7 to 1.0. Therefore, for an elliptical wing the value is 1.0 and can be omitted from the formula, this being the most efficient planform where the lift coefficient remains the same all along the span. Rectangular shaped wings are the least efficient planform and have a value of approximately 0.7, due to increased tip vortices. Planforms with taper and/or washout use intermediate values between 0.7 and 1.0, due to their improved efficiency over a rectangular planform wing.

In 1917, Professor Ludwig Prandtl (1875–1953) discovered the induced drag is proportional to the aspect ratio.

See **Elliptical wing planform; Span efficiency factor; Diagram 20, Total Drag Curves.**

**Induced inflow & angle** The induced inflow is the low-pressure area of the slipstream drawn into the front of the propeller disc, or from the rear during reverse thrust operations.

On a helicopter, the induced inflow is the air drawn in through the top of the rotor disc and is forced downwards by the rotor blades during normal flight operations. It enters from below the rotor disc during autorotation manoeuvres and when flaring to land. The normal inflow direction on an autogyro is from below the rotor disc.

**Diagram 44, Induced Inflow & Angle**, shows the rotor RPM or rotational velocity ( $V_R$ ) vector as A-B, which is assumed to be a constant RPM. The induced inflow, vector C-B represents the air drawn in through the top of the rotor disc, which is a variable within limits. Therefore, the resultant vector due to the RPM and induced inflow is vector A-C known as the relative airflow.

The inflow angle, which is also a variable, is dependent on the induced inflow velocity vectors C-B. The angle of attack is the angle between the rotor blade's chord line and the relative airflow (RAF). Any increase in the induced inflow velocity will reduce the angle of attack of the rotor blades. From the above we can conclude that:

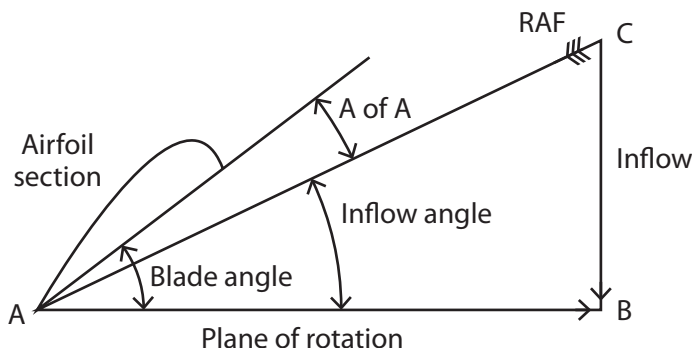


- the RPM and inflow angle are inversely proportional for any given induced inflow velocity
- the induced inflow velocity and angle of attack are inversely proportional for a given rotor RPM ( $V_R$ ) and blade pitch angle.
- The induced inflow and inflow angle is proportional to the prop/rotor RPM.

Some parts of the induced flow pass outside of the rotor disc. Due to the helicopter's forward speed, the airflow approaches the rotor disc horizontally until being drawn downwards through the disc; however, the helicopter will have moved forward passed a portion of the induced flow, which does not enter the rotor disc.

**Diagram 45, Inflow (Transverse) Roll**, also can be used to show the effects of changes in the inflow vector. Note for a helicopter, the vector C-B represents the induced inflow air entering the rotor disc from above; on a propeller, the inflow vector C-B represents the inflow velocity due to the aircraft's forward speed. This diagram is similar to **Diagram 68, Propeller Pitch**.

The term induced inflow can also be applied to the air drawn into an engine's intake.



**Diagram 44, Induced Inflow & Angle**

**Induced outflow** The induced outflow, obviously is the opposite of the induced inflow. It is the high-pressure area of slipstream expelled behind the propeller or through the underside of a helicopter's rotor disc during normal flight.

**Induced power** The engine's induced power is proportional to the lift times the vertical velocity of the downwash from the wing's trailing edge. The induced power is related to lift and increases as the load squared and varies as  $1/\text{speed}$ , reducing with increasing aircraft speed.

The induced power is the power required to drive the helicopter's main rotor disc to provide the total rotor thrust (lift) to equal the helicopter's weight. The induced power reduces with increasing speed.

*See* **Power curves – helicopter.**

**Induced velocity** The induced velocity is the speed of the airflow over the wing induced by the downwash and trailing vortex system. The induced velocity is proportional to the lift generated by the wing.

**Inertia** Inertia is the Latin word for inaction. It can also be described as a state of inactivity or rest of a body (aircraft). Inertia is the tendency for a body to remain at rest or to continue in uniform motion in a straight line unless disturbed by some other external force.

Inertia depends on the mass of the body, but not its weight (an increase in altitude will affect the inertia) where a greater mass requires a greater inertia to change its motion. For example, an aircraft will remain stationary until power (thrust) is applied to start it moving. When flying straight and level at constant speed, it continues to do so until a control input or power change alters its flight path.

Inertia should not be confused with momentum. With an increase in air speed, momentum increases while the inertia remains constant.

Galileo Galilee introduced the concept of inertia, which was further developed by Newton as part of his First Law of Motion. It is also known as the Law of Continuity.

**Inertia axis** Fighter aircraft have the engine(s), crew and weapon stores, etc, mounted in or, along the fuselage length. This weight can be considered as two masses in front of and to the rear of the actual centre of gravity. The inertia axis acts through these two masses and the CG, in the longitudinal direction.

**Inertia coupling** Inertia coupling is a dangerous flight condition – an instability problem – that beset some early jet fighter aircraft.

A flight upset can occur during high-speed flight manoeuvres especially in the rolling plane. Compared to the wings, the fuselage being relatively heavy causes the problem (the fuselage holds the crew, engines, electronic equipment, etc). The wings are relatively lightweight, short span with low roll inertia, compared to the high pitch and yaw inertia of the fuselage. The result becomes apparent in rolling manoeuvres. The aircraft may be rolled with a high rate of roll ( $400^\circ/\text{second}$  is possible with some fighters). Inertia forces in the roll can overpower the aerodynamic stability forces inherent in the wings and fuselage. The fast rate of roll induces the fuselage to pitch and yaw of its own accord. The resultant oscillation can be too great for the pilot to control leading to total loss of control of the aircraft.

The aerodynamic OX longitudinal axis acts through the centre of gravity. If the aerodynamic and inertia axis are in alignment then inertia coupling will not be present if the aircraft rolls. However, during manoeuvres with the inertia axis inclined above the OX longitudinal axis, inertia forces tend to dominate the stabilizing aerodynamic forces to create a centrifugal force and a high rate



**Photo:** A North American F-100 Super Sabre was the first victim to suffer the effects of inertia coupling. This F-100 survives at the Pima Air & Space Museum at Tucson, USA.

of roll with adverse yaw, which induces uncontrollable pitch oscillations around the inertia axis; this is inertia coupling.

The opposite effect occurs when the inertia axis is below the OX longitudinal axis and inertia-coupling causes proverse yaw as opposed to adverse yaw when the inertia axis is above the OX longitudinal. Inertia coupling may be a problem when the inertia axis is located either above or below the OX longitudinal axis. At increased altitudes, the inertia axis is usually located above the OX longitudinal axis due to the less dense air and inertia coupling is a greater possibility.

The North American F-100 Super Sabre was the first aircraft to suffer from inertia coupling and a production model was used to investigate the cause of the instability upset. The cause of inertia

coupling was explained mathematically in 1948 by English born, NACA engineer (William) Hewitt Phillips (1918–2009).

Inertial coupling is also known as roll coupling, or simply as, coupling.

*See* **Inertia axis.**

**Inertia stall** At altitudes above 40000 feet, the speed range of jet transports reduces considerably between the low-speed aerodynamic stall and high-speed Mach buffet stall. Encountering turbulence or rough pilot handling can induce control loss due to compressibility effects in the high subsonic speeds. In addition, any sudden change in aircraft pitch attitude produces a gradual change in the aircraft's flight path, which increases the angle of attack until the low-speed aerodynamic stall speed is reached.

*See* **Coffin corner.**

**Infinite aspect ratio** Aerodynamicists, when working on airfoil calculations may choose to ignore the effects of the airflow around the wingtips. When ignoring wingtip effect, it is assumed the wing is of an infinite aspect ratio, that is, it has no end. Therefore, the results of the calculations would not be the same as for a finite wing and would have to be adjusted accordingly.

**Infinite wing** An airplane with an infinite wing is physically impossible. However, an infinite wing can be considered as a wing under test in a wind tunnel, where it stretches from wall to wall preventing the formation of wingtip vortices; this can alter the data collected from the tests giving incomplete results and therefore, this requires a correction factor. The lift curve slope is steeper for an infinite wing than a comparable finite wing.

*Compare with* **Infinite aspect ratio; Finite wing.**

**Inflow angle** The inflow angle is the angle made between the plane of rotation (angle A-B on *Diagram 44, Induced Inflow & Angle*) and the relative airflow (A-C). The angle will increase with increasing forward speed/inflow velocity and propeller or rotor blade RPM. The vector C-B. represents the inflow velocity.

See **Induced inflow & angle.**

**Inflow factor** The inflow factor is the ratio of increase of slipstream velocity through the propeller disc to aircraft velocity.

The slipstream velocity/aircraft velocity ( $v/V$ ) ratio is known as the inflow factor, which increases with an increase of slipstream velocity. Propeller efficiency will be greatest when the slipstream velocity ( $v$ ) is close to the aircraft's true air speed ( $V$ ) that is, greatest efficiency is achieved at a small value of  $v/V$ . When the slipstream velocity equals the aircraft's velocity, the value of the inflow factor is equal to one. At other speeds, where the slipstream velocity is less than the aircraft's velocity (which is the usual case) the value of the inflow factor will of course, be less than one. This ties in with the ideal efficiency of the propeller, where the thrust is related to the aircraft's speed and the inflow factor. The ideal efficiency or Froude's Efficiency can be calculated from the formula known as Froude's Equation:

$$\text{Ideal efficiency} = \frac{T \times V}{T \times (V + v)}$$

Where  $T$  = thrust factor

$V$  = aircraft velocity

$v$  = slipstream velocity.

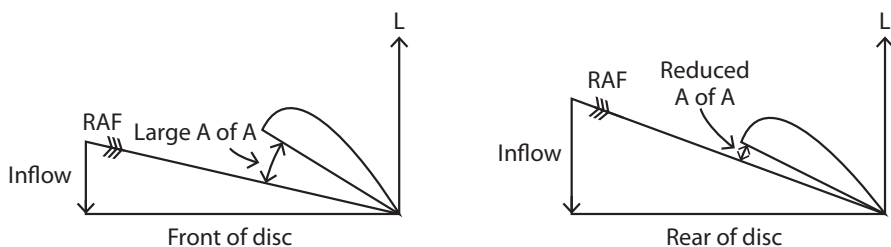
See **Propwash-thrust.**

**Inflow roll** A helicopter can experience an uncommanded roll, usually to the right, due to a non-uniform velocity flow across the rotor disc.

During forward flight, the airflow at the rear of the rotor disc has a higher downwash inflow velocity than that at the front of disc; this is due to its increased acceleration of air as it passes across the disc. The downwash inflow of air with increased velocity enters the rear of the disc at a reduced angle of attack and hence reduced lift (rotor thrust) and drag, as opposed to greater lift and drag at the front of the disc. The difference in lift and drag across the rotor disc coupled with gyroscopic precession causes the disc to tilt towards the right (assuming a counter-clockwise rotation viewed from above). The greater lift force at the front of the disc is translated  $90^\circ$  around the disc to raise the rotor blades on the left hand side of the disc with a corresponding lowering on the right side (the blades flap up and down). Coupled with the flap-back, the helicopter tends to pitch up and roll to the right.

Inflow roll is more apparent at low forward speed near 10 to 20 Knots. Additionally, due to the coning angle meeting the relative airflow at an increased angle of attack at the front of the disc and a reduced angle of attack at the rear, it results in a rolling moment as forward speed increases.

This is also known as transverse roll.



**Diagram 45, Inflow (Transverse) Roll**

**Inflow velocity** As the propeller rotates under normal operating conditions, it sucks air from in front of the propeller disc causing a low-pressure area. The air mass known as the induced inflow accelerates through the propeller disc into an area of increasing velocity behind the propeller and experiences a rapid rise in pressure as it does so. It is the difference in pressure between the front and rear of the propeller disc, caused by the change in the air mass momentum that produces thrust. The air mass now called the outflow velocity continues to accelerate reaching its maximum velocity some distance behind the propeller. The final maximum velocity is equal to the aircraft's true air speed plus double the slipstream velocity.

It should now be obvious from the above, half of the slipstream velocity increase occurs in front of the prop while the other half of the speed increase occurs behind the prop. As the speed of the slipstream increases to its maximum value, the slipstream also contracts to a smaller diameter than the prop disc itself, in compliance with Bernoulli's Theorem. The point of constriction is known as the Vena Contracta. The air mass flowing through the prop disc should be considered as a three dimensional stream tube.

*See Induced inflow & angle; Propwash-thrust.*

**In ground effect (IGE)** A helicopter is said to be 'in ground effect' when it is hovering within one rotor disc distance above the ground.

*See Ground effect.*

**Inherent stability** Inherent stability is an airplane's natural tendency to return to its in-flight trimmed condition after a disturbance. Stability is not automatic but depends on aircraft design and response to manoeuvres and upset.

*See Stability.*



**Inlet cones** Inlets cones are located at the entrance to a supersonic jet engine's inlet. They are of a conical shape and designed to induce a conical shockwave from the cone's apex. The combined inlet cone and shockwave reduces the intake velocity from a supersonic flow to a subsonic speed, before entering the jet engine. The inlet cone's position is automatically variable within the engine intake, translating aft with increasing airflow velocity to match the conical shockwave's reducing cone angle.

The inlet cone has an improved boundary layer airflow and better internal compression within the conical flow, which reduces flow separation and improves pressure (ram) recovery for the intake's inflow.

Inlet cones are also known as inlet centre body or shock cones.



**Photo:** The Lockheed SR71 Blackbird has twin inlet cones, essential for supersonic flight. This one is on display at the Pima Air & Space Museum in Tucson, AR.

**Inset hinge** On a primary control surface (rudder, elevator or aileron) the inset hinge is located aft of its leading edge. Its purpose is to assist the movement of the control surface when activated by the pilot. The part of the surface ahead of the hinge line deflects into the slipstream increasing the camber and the velocity of the air flowing over it, which increases its lift and moves the centre of pressure closer to the hinge line making the control easier to move.

It is similar to a horn balance in its action.

**Inside wing** An airplane's inside wing is that wing nearest to the centre of the turn.

*See* **Outside wing**.

**Inspin yaw** For an airplane to enter a spin, a stall and a yaw condition is required. The yawing moment is known as an inspin yaw; it induces the airplane to enter the spin and to hold it in a spin once it is developed. Preventing a yaw when the airplane stalls will prevent a spin.

**Instability** Instability is the tendency of an airplane to diverge from its intended flight path following a disturbance.

**Institute of Aerodynamics, Aachen** Professor H. Reisner founded the Institute of Aerodynamics at the University in Aachen, Germany in 1912.

Four years earlier, in 1909, Professor H. Reisner with Hugo Junkers (1859–1935) of Junker's airplane fame had already started fluid flow experiments. Together they carried out major theoretical research into flight stability and airplane manoeuvrability.

By 1913, Professor H. Reisner had left his position and was replaced by Professor Theodore von Karman (1881–1963) who held the chair of mechanics and aerodynamics until 1930. Von Karman then moved to the Guggenheim Aeronautical Laboratory of Caltech in the USA, while staying associated with the Aerodynamic Institute at Aachen.

**Intake momentum drag (Ram drag)** Subsonic jet aircraft have their gas turbine engine's intake formed in the shape of a divergent duct to reduce the intake velocity and increase the air pressure before entering the compressor. At the start of the take-off run, the drop in air pressure in the intake may be seen when visible moisture is formed, especially on high-humidity days. Initially, the compressor is sucking air in through the intake causing the small drop in pressure but as the aircraft's speed increase during take-off and climb, ram effect/recovery takes over and the air then enters the intake at an increased pressure (and relatively lower velocity).

The difference between the engine's gross thrust and net thrust is caused by the intake momentum drag. When the aircraft is stationary with the engine idling, net thrust and gross thrust are equal; with increasing forward speed, gross thrust exceeds the net thrust due to ram recovery.

See **Convergent duct; Divergent intake duct; Ram effect/recovery; Venturi.**



**Photo:** A North American F-86 Sabre has a single subsonic intake. Photographed at Pima Air & Space Museum, Tucson, Ar.

**Interference** Interference refers to the interaction of the aerodynamic forces acting on the upper and lower wings of a biplane. The dimensionless Prandtl-Glauert interference factor is used to determine the amount of interference present between a biplane's wings.

*See* **Biplane interference.**

**Interference drag** The conflicting pressure fields over two adjacent parts of the airframe cause interference drag.

Areas such as the wing roots and fuselage junction on low wing aircraft are particularly prone to interference drag caused by the mixing or interference of the boundary layer; hence the need for fillets to reduce the drag as much as possible. Interference drag (a portion of parasite drag) also manifests itself between the aft fuselage and rear fuselage-mounted jet engines and on T-tail aircraft, etc. Interference drag has two causes: in the first instance, drag is increased where a bulging surface accelerates the passing airflow and increases the skin friction drag on adjacent surfaces. In addition, the second cause is due to the airflow slowing down through a constricted space, which can cause flow separation and turbulence.

Jean Charles Borda (1733–1799) a French physicist, engineer and mathematician, was the first to recognize the interference between conflicting air flows.

*See* **Area rule; Drag; Fillets.**

**Internal aerodynamics** The study of airflow through an enclosed channel is known as internal aerodynamics. It includes flows through venturi tubes, convergent or divergent ducts and jet engines, for example.

**International Standard Atmosphere** The Earth's maximum temperature of 57.4°C (136°F) was recorded at El Azizia, near Tripoli in Libya in 13 September 1922 and the lowest, reliable minimum recording of -89.2°C (-128.6°F) was recorded at the Russian base of Vostok, Antarctica in 1997. Between these two temperature extremes, the temperature varies at any given place on Earth on a diurnal, daily, and seasonal basis. The temperature also decreases with an increase in altitude and with an increase in latitude from the equator to the poles. From the General Gas Laws we know that a change in temperature also produces a change in pressure and density/volume of the air. It follows, temperature, pressure and density in the atmosphere varies considerably from day to day.

For the purpose of aircraft design and performance comparison and the graduation of altimeters, etc, several average conditions are considered and used as a basis for reference purposes. ICAO adopted the International Standard Atmosphere (ISA) to be the official reference for the aviation industry. The ISA conditions at Sea level assuming dry air conditions (water vapor 0% to 4%) are as follows:

- a temperature of 15°C (59°F)
- a temperature lapse rate of 1.98°C (3.5°F) per 1000 feet up to a height of 36090 feet where the temperature is assumed to remain constant at -56.5°C up to 65600 feet
- a pressure of 1013.25 hPa (Hectopascals) =  $101.3 \text{ kN/m}^2$  = 760 mm hg. = 29.92 inches of mercury (in. Hg)
- a density of 1.225 kg/m<sup>3</sup> (0.0765 lbs/cu.ft).
- The air is dry.

The American, Dr. Willis Ray Gregg (1880–1938) introduced the term standard atmosphere in 1922.

See **Atmosphere; Structure & composition of the atmosphere.**

**In track** A helicopter's rotor blades must remain in track to ensure smooth running. The blades are in track when all tips pass through the same plane as they rotate.

**In trail** A control surface is in trail when it is aligned with the fixed surface ahead of it, to which it is hinged. When in trail, the drag is less than when the control surface is deflected. All-flying tailplanes do not have this problem.

**Inverted gull wing** Inverted gull wings are in the shape of a capital 'W'. From the wing root, the wing slopes down (anhedral) and then angles upwards (dihedral) towards the tips.

The WWII Chance Vought F4U Corsair must be one of the most famous airplanes of all time with inverted gull wings. The reason for the inverted gull wings on the F4U Corsair is due to the ground



**Photo:** Its inverted gull wings easily identify the Chance Vought F4U Corsair of World War II fame. The prototype made its first flight on 29 May 1940 and over 12000 were built during World War II. This photo shows a Goodyear-built Corsair FG-1D model.

clearance necessary for the large radius propeller (the largest on any fighter). The inverted gull wings have the undercarriage mounted on the wing's low point. It first flew on 29 May 1940. Another aircraft with gull wings that first flew in 1935 was the German Junkers JU 87 Stuka dive-bomber. [It also had flaperons]. The British Royal Navy's Fairey Gannett, anti-submarine aircraft also had inverted gull wings. It first flew on 19 September 1949.

It is also known as a cranked wing.

*Compare with* **Gull wing**.

**Inverted spin** An inverted spin occurs, as the name implies, when the aircraft is inverted (upside-down). For the pilot, it can cause discomfort and confusion in the required control input for recovery. The direction of roll and yaw are opposite to a normal spin condition. Recovery requires the movement of up-elevator to recover (the airplane is inverted) not down-elevator as in a normal spin. In addition, just to confuse matters even more for the pilot, elevator reversal can also occur.

**Inverted stabilizer** A helicopter's stabilizer located on the tail boom may be an inverted cambered airfoil; the manufacturer's flight-tests determine the required location of the stabilizer, which depends on the effects of the rotor downwash and helicopter forward speed. The stabilizer provides a down-force on the rear of the helicopter and damps out dynamic oscillations of the fuselage and rotor disc due to wind gusts or rotor disc blowback, which causes a pendulum motion of the fuselage and rotor disc. When the helicopter accelerates into forward flight, the rotor disc tips forward and the helicopter's fuselage follows suit until equilibrium of the forces exists. At this position, the sum of all forces and moments will be zero. The fuselage nose-down moment is balanced by the stabilizer's tail-down force. Stabilizers are essential for stability in forward flight but during hovering flight, they have no effect on stability. When hovering rearwards the longitudinal

stability is reduced. However, stabilizers extend the helicopter's centre of gravity range.

See **Stabilizers**.



**Photo:** The Aerospatiale AS-350 Squirrel has a prominent inverted stabilizer to produce a down force on the tail during forward flight to damp out any dynamic oscillations of the fuselage and rotor disc. Compare the airfoil section shape and location with the photo of the NH 90's stabilizer (at entry for Gurney flap).

**Inviscid flow** Inviscid flow describes a fluid flow as one without viscosity, that is, a flow devoid of any internal friction or drag force. In reality, all flows are viscous to a certain degree. The airflow within the boundary layer exhibits the full effect of a viscous flow as opposed to an inviscid airflow as found outside of the wing's boundary layer.

The idea of drag being absent in an inviscid flow is known as d'Alembert's paradox, which he introduced in 1744.

See **Diagram 89, Transition Point & Region** and compare with **Viscous flow**.



**Irreversible flow** An irreversible flow (a thermodynamic term) is a system where finite changes occur in a supersonic flow where the system is not in equilibrium, and the entropy increases. An irreversible flow is an opposite process to an isentropic flow.

Normal or oblique shock waves are formed where a supersonic flow is abruptly turned into a decreasing area, and the flow becomes irreversible and a non-adiabatic process.

*Compare with Isentropic flow.*

**Irrotational flow** The individual fluid element's movement is all translational with no angular rotation about its axis. Vorticity is zero everywhere throughout the flow. It is also known as potential flow.

*See Potential flow.*

**Isentropic flow** The term isentropic is a thermodynamic expression meaning without change in entropy (a measure of the energy that is unavailable to do work). An isentropic process deals with the relationship of pressure and density. It is also an adiabatic and reversible process and therefore it maintains constant entropy; it follows an irreversible isentropic process is not adiabatic; this is stated in the second law of thermodynamics. A gradual change in the flow conditions through a convergent-divergent duct produces an isentropic flow. A supersonic flow through an expansion flow (with increasing area) is also isentropic, known as an isentropic expansion, but not so through a convergent flow shock wave (with decreasing area) where the flow becomes irreversible. The generation of sound waves is also isentropic.

The term isentropic is taken from the Greek word 'iso' meaning 'the same' and the word entropy.

*See Thermodynamics; Isobaric; Adiabatic process; Irreversible flow; Open cycle; Entropy; Isothermal.*

**Isobaric** An isobaric process is one where the gas (air) pressure is maintained constant but its volume is free to expand and contract and any pressure changes due to heat transfer would be neutralized.

*See* **Thermodynamics; Isentropic flow; Adiabatic process; Open cycle; Entropy; Isothermal.**

**Isothermal** If the temperature remains constant in a thermodynamic process, it is said to be isothermal. During an isothermal process for an ideal gas, if the temperature remains constant the change in internal energy of the system remains constant.

*See* **Thermodynamics; Isentropic flow; Adiabatic process; Open cycle; Isobaric.**

## J

**Jacobs, E.** Eastman N. Jacobs (1902–1987) was a leading aerodynamicist employed by NACA Langley Aeronautical Laboratory in the 1920s and 30s and he became the head of the variable-density wind tunnel division from 1928 to 1939. He was instrumental in researching high-performance airfoils in the variable-density wind tunnel, studying boundary layer turbulence, which continued on to the development of laminar flow airfoils in the NACA 4-digit series, with the help of Schlieren photography. Jacobs noted that flow separation was delayed on laminar flow airfoils and it was not the actual shape of the airfoil but the pressure distribution about the airfoil, which affected flow separation. The success of the North American Mustang and other fighter aircraft with laminar flow airfoils was due to his research.

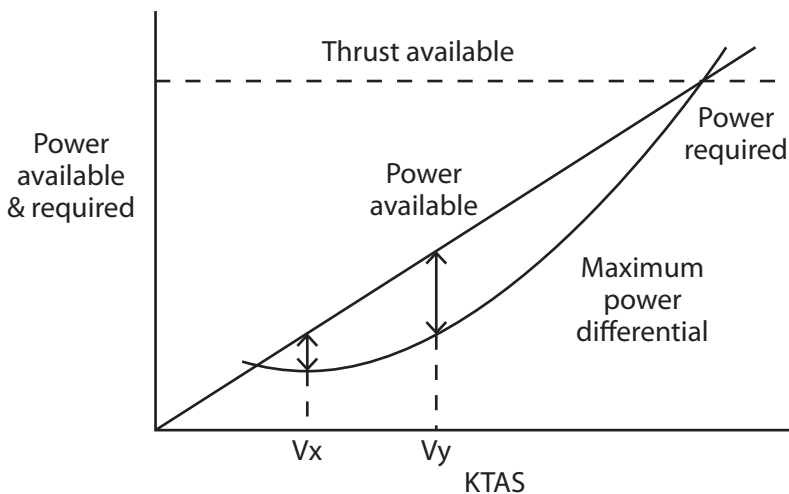
*See Separation point & flow.*

**Jet-engine power** *Diagram 46, Jet-engine Power Curves*, shows the curves for power available and power required for a jet engine. The power available (determined by the engine's output) is calculated from thrust times true air speed for selected speeds over the aircraft's speed range. Jet power available varies only by a little amount over the aircraft's speed range as indicated by the straight line. [The thrust horsepower available for a piston-engine is variable and therefore curved]. On the other hand, the power required curve is similar to the piston-engine's thrust horsepower available curve, because thrust required is equal to drag. The aerodynamics of the aircraft determine the power or thrust required. The maximum rate of climb speed ( $V_y$ ) is located at the maximum power differential (shown between the power available and required curves) where the excess power is achieved. The speed range for climbing is greater than that for a piston-engine aircraft. The maximum angle of climb speed ( $V_x$ ) occurs at the minimum drag speed.

The power available decreases with altitude (the power available straight line curve would be angled down as a less steep slope). The power required curve moves up the graph and to the right (the same as a piston-engine power curve) with increasing altitude. The maximum excess power decreases with a corresponding decrease in the rate of climb. The maximum rate of climb speed (TAS) increases with altitude but the indicated air speed (IAS) decreases in accordance with the IAS/TAS ratio.

Jet-engine power is commonly measured in pounds of thrust required and thrust available, but less often, reference is made to power available and power required, which is measured in horsepower. **Diagram 46, Jet-engine Power Curves**, shows this.

*Compare with* **Piston-engine power curves – airplane.**



**Diagram 46, Jet-engine Power Curves**

**Jet flaps** Jet flaps is an English term for a boundary layer control device, which uses forced air to flow over the flaps to increase their effectiveness. The Americans refer to them as blown flaps.

See **Blown flap**.

**Jet upset** *See Coffin corner.*

**Joined wing** *See Diamond wings.*

**Joukowski, N. Prof** Professor Nikolai Yegeorovitch Joukowski (1847–1921) was born in Orekhova, Central Russia. He became a professor of Mechanics at Moscow University and was Russia's top aerodynamicist. Like Martin Wilhelm Kutta (1867–1944), he also worked on the problems of lift, airfoil sections, and circulation. Joukowski also developed Russia's first wind tunnel circa 1904 at the Moscow Higher Technical School, the world's first aeronautical institute, at Kachino near Moscow. He was the founder of the Central Aerodynamics & Hydrodynamics Institute, which was established on 1 December 1918.

He was also known, more commonly, as Professor Nikolai Zhukovski, the English spelling for his Russian name. He became Russia's most famous aerodynamicist earning the title of the 'Father of Russian aviation'.

**Joule (J)** The SI unit of energy, known as the Joule, which represents a force of one Newton-meter (N-m). The unit Joule was named for James Prescott Joule (1818–1889).

*See Newton (force).*

**Joule's cycle** *See Brayton cycle.*

**Judder** Judder is an alternate name for buffet, caused by the turbulent airflow flowing off the main wing and over the tailplane at relatively high angles of attack, prior to the wing stalling.

**Junkers flaps** Unlike conventional control surfaces, the Junkers flap is located below the trailing edge of the main wing. The space between the flap and wing under-surface creates a venturi effect where the airflow increases velocity and re-energizes the

airflow boundary layer over the wing's upper surface, especially at relatively low air speeds. This makes the Junkers flap ideal for STOL type aircraft.

Normal type ailerons attached to the trailing edge of the wing, operate in the thickened turbulent boundary layer where greater control deflection is required to produce a given aircraft movement about its axis. Junkers flaps outside of the boundary layer's influence are more effective and produce a greater degree of aircraft movement for the same given control deflection. One disadvantage of the Junkers flap is they are less efficient at higher speeds due to the venturi effect causing drag and sapping energy from the airflow.

If the Junkers flap also doubles as ailerons, it is then known as a flaperon. A German engineer, Otto Mader (1880–1944) at Junkers, developed it.



**Photo:** The Junkers flaps are used on the three-engine Junkers Ju 52/3m. This aircraft is located at the National Museum of the US Air Force in Dayton, Ohio.

# K

**'K'** The symbol represents a factor of approximately one point one. It is applied to calculations for a given wing to show by how much its induced drag is greater than that of a wing with a perfect elliptical lift distribution.

*See Induced drag.*

**Karman, T.** Professor Theodore von Karman (1881–1963) was born in Budapest, Hungary. He studied initially in Budapest before joining Professor Prandtl (1880–1953) at Gottingen, Germany in 1906 where he wrote his PhD.

He later became a professor and Director of Aerodynamics at the Institute of Aerodynamics in Aachen, Germany. He commenced his work in fluid dynamics in 1912 and he is noted for his studies in the laminar and turbulent boundary layer theories and high subsonic, transonic and supersonic flows, the Karman Vortex Street, JATO rockets, guided missiles and jet propulsion, etc. In 1932, he developed the Karman-Moore aerodynamic theorem of supersonic flight for slender body aircraft. In 1939–41, he worked on the effect of compressibility and the variation of the centre of pressure with increasing Mach number. This resulted in the Karman-Tsein theory.

Von Karman moved in 1930, to Pasadena, California, where he became the Director of the graduate aerodynamics laboratory at the Guggenheim Aeronautical Laboratory of Cal Tech. He commuted regularly to the Institute of Aerodynamics at Aachen, Germany. He is considered the father of supersonic flight and with a colleague, he coined the term 'supersonic'.

Von Karman was the founder of the NATO Advisory Group for Aeronautical Research and Development (after WWII) and in 1963, he received the first US National Medal of Science presented by President Kennedy. After receiving this medal, he died shortly after at Aachen, Germany in 1963.

This is a brief description on what Theodore von Karman has achieved – his resume is outstanding!

**Karman-Moore theory** In conjunction with Norton B. Moore in 1937, Professor Theodore von Karman (1881–1963) developed the aerodynamic theory for slender body airplanes flying at supersonic speeds.

**Karman-Tsein theory** Professor Theodore Von Karman (1881–1963) and associate Hsue-shen Tsein (1911–2009) developed their theory of the effects of compressibility and the variation of the centre of pressure on airfoils with increasing Mach number in 1939–41. This was known as the Karman-Tsein theory.

*See Tsein H..*

**Karman vortex street** Professor Theodore von Karman (1881–1963) discovered the vortex street phenomenon occurs in highly unstable boundary layer separation.

The name Karman vortex street describes the vortex shedding sequence in the wake as a row of staggered vortices, alternating anti-clockwise and clockwise behind a bluff body located in the flow; low-pressure drag is produced at the stagnation point behind the body. With increasing Reynolds number  $Re$  40 to  $10^2$ , the vortices become continuous, as they are washed downstream. This accounts for the noise of ‘whistling in the wires’, familiar to sailors and pilots of biplanes. The vortex street can also be found in cloud formations and flowing water.

It is also known as vortex shedding, or vortex trail.

**Keel effect** The term keel effect is an alternative name for pendulum stability. It is more pronounced on high wing aircraft than low wing types due to the greater vertical distance between the centre of lift and the aircraft’s centre of gravity. The large restoring moment induced due to a sideslip helps to bring the aircraft back to wings level flight.

*See Pendulum stability.*



**Kelvin's circulation theorem** An involved and very complex theorem of circulation and lift production, which leads to the starting vortex theory. William Thomson, 1<sup>st</sup> Baron, Lord Kelvin (1824–1907), an Irish born mathematical physicist, noted for his work in thermodynamics and aerodynamics introduced this theorem in 1869.

See **Starting vortex; Circulation theory of lift ( $\Gamma$ )**.

**Kinematic viscosity** Kinematic viscosity determines a fluid's resistance to flow or its degree of stickiness. It is measured by the fluid's viscosity divided by the fluid's density, which varies with temperature and pressure. Kinematic viscosity is a dimensionless number.

The kinematic viscosity is shown by the following formula:

$$\text{Kinematic viscosity} = \nu = \mu/\rho$$

Where  $\nu$  = kinematic viscosity

$\mu$  = fluid viscosity

$\rho$  = fluid density.

**Kinetic energy** The term kinetic energy refers to a mass of  $m$  kilograms traveling at a speed of  $V$  meters per second squared.

The kinetic energy of one centimeter of air of a given density ( $\rho$ ) and moving at a given velocity  $m/s^2$  is expressed as  $\frac{1}{2} m V^2$  Joules. When converted to pressure energy and brought to rest it is called dynamic pressure ( $q$ ), which equals  $\frac{1}{2} \rho V^2$ . The mass ( $m$ ) is changed to air density ( $\rho$ ) when associated with dynamic pressure, as found in the lift and drag formulas for example.

Doubling the air speed of an aircraft will quadruple its kinetic energy. Dynamic pressure plus static pressure equals a constant.

See **Energy – potential & kinetic; Bernoulli's theorem; Conservation of Energy**.

**Kinetic heating** Kinetic heating is due to the skin friction and compression of the air as it flows over the aircraft. The heating increases with the square of the speed.

Kinetic heating is insignificant at the speeds of light aircraft, the temperature rise being about 2°C at 150 Knots true air speed. However, it becomes of greater significance as Mach 1.0 is approached and exceeded. Concorde cruising at Mach 2.0 has a temperature rise of up to 180°C and expands about 15 centimeters (six inches) during cruise flight due to kinetic heating! High-speed aircraft require titanium alloy or other heat resistant materials for their outer surface to withstand the high temperatures generated by kinetic heating. The Space Shuttle has a temperature rise of approximately 1500°C during re-entry from space, hence the need for special heat resistant tiles on the fuselage and wings.

Temperature rise in degrees Centigrade = (TAS MPH/100)<sup>2</sup>.

See **Aerodynamic heating; Temperature rise.**

**Kinetic pressure** See **Dynamic pressure.**

**Kinks** The Avro Vulcan delta wing bomber (first flight 1952) had a triangular shaped addition forming a kink on its outer leading edge, to improve handling problems.

The English Electric Lightning fighter with highly swept wings also had leading edge kinks on its outer wing panels.

**Knudsen number** The ratio of the molecular mean free path to a given length, is defined by the Knudsen number (Kn). The dimensionless Knudsen number is defined as:

$$Kn = \lambda/L$$

Where  $\lambda$  = molecular mean free path

L = the physical length.

Aerodynamicists may use the Knudsen number in their calculations of the airflow over an aircraft in high-speed flight.

The Danish physicist Martin Knudsen (1871–1949) introduced the ratio named after him, for use in statistical or continuum mechanics. It is related to Mach and Reynolds numbers in statistical or continuum mechanics.

**Krueger flaps** A Krueger flap is a leading edge device found on the inboard leading edge of Boeing jet transport aircraft. They have the effect of lowering the stagnation point on the leading edge, which directs more air over the wing's upper surface to increase the energy in the boundary layer.

The Krueger flaps are stored in the wing's lower surface near the leading edge and are hinged to swing downwards and forwards into position for take-off and landing and are sometimes referred to as 'droops'. The Krueger's provide a bluff leading edge on the main wings to induce greater airflow over the upper surface to increase lift. Unlike leading edge flaps, they do not create a slot at the leading edge but they do create a larger camber to the airfoil. They are more complex to build than leading edge flaps.

They were invented by the German aeronautical engineer, Werner Krüger (1910–2003) in 1943 and tested at the Gottingen Aeronautical Research Institute, Germany. The Boeing Company of America flight-tested Krueger flaps on their Boeing Model 368-80 (707 prototype) which first flew on 15 July 1954. The Boeing 727 was the first production aircraft to use them, making its maiden flight on 9 February 1963.

Note spelling of Krueger flaps and Werner Krüger's name.

See **Slats**; **Slots**; **Powered slat**.

**KTAS** The true air speed of aircraft is now commonly measured in Knots. One Knot is equal to 1.15 MPH or 1.85 KMH. The word Knots represents the speed for 'nautical miles per hour', although it is incorrect to use the full term.

**Küchemann, Dr. D.** Dr. Dietrich Küchemann (1911–1976) was a prominent WWII scientist and aeronautical engineer, born in Gottingen, Germany.

Dr. Küchemann is famed for his early research into supersonic aerodynamics and later (1942 to 1945) for being a wind tunnel expert at the Volkenrode Aeronautical Research Institute in Braunschweig, Germany. At Volkenrode, he conducted research into high-speed flight, specializing in the advanced study of swept wing aircraft and shockwave drag, etc. He was also making progress towards discovering the area rule concept to reduce the spanwise flow on swept wing high-speed aircraft. [However, the discovery of area rule to reduce drag has been credited to Richard T. Whitcomb (1921–2009) at NASA Langley Research Centre]. Dr. Küchemann joined Professor Ludwig Prandtl in aerodynamic research and gained a Doctorate in 1936.

Dr. Küchemann worked at the Royal Aircraft Establishment in England as an aeronautical scientist from 1946 until 1976. Here, he specialized in aircraft propulsion becoming a leading authority on ducted fans and the installation of jet engines. He also conducted research into lifting-body aircraft and flow separation over wings. He played a leading role in the development of thin-wing delta aircraft and design work for Concorde's Ogive shaped wing in conjunction with fellow Germans, Eric Maskell and Dr. Johanna Weber.

In 1954, he became the Senior Principle Officer at the Royal Aircraft Establishment and in 1966, the Chief Scientific Officer of the aeronautics department (a post held in earlier years by Hermann Glauert until his death in 1934). In 1964, he received the well-earned CBE (Commander of the British Empire) before retiring in 1971, only to continue part-time until 1976.

**Küchemann carrots** Küchemann carrots are pods or anti-shock bodies placed on the wing's trailing edge of high subsonic aircraft – an invention of Dr. Dietrich Küchemann (1911–1976).

The carrots reduce the strength of shockwave drag that may form preventing flow separation and reducing the drag rise associated with transonic speed. They also save fuel (less drag means less thrust required and therefore less fuel). The carrots act in a similar manner to area rule.

Also known as speed bumps.

See **Shock body**; **Flap track fairings**.

**Küchemann tips** Küchemann wingtips were developed early in the jet age – 1950s to 1960s – for jet airliners. They are characterized by having well-rounded wingtip leading edges. From a drag point of view, they were not ideal. However, their big advantage is at high



**Photo:** An early jet-age aircraft, the Vickers VC-10 had distinctively shaped Küchemann wingtips to improve aerodynamic performance.

cruise Mach numbers where the tips delay shock wave formation and increase the drag rise Mach number ( $M_{DR}$ ) and improve the wingtip lift.

A type of streamwise tip.

**Kutney** The tapered Kutney bump is a drag-reducing bulge on the side of the inboard jet engine pylon. They are shaped as a tapered, outward-bulging non-symmetrical contoured pylon fairing.

A conventionally shaped jet engine pylon on a jet transport aircraft has a maximum thickness about 20–40% MAC similar to a vertical, symmetrical stabilizer. The fuselage, under wing area, engine and pylon form a channel through which, the airflow may reach supersonic speed. The shockwave formed in the channel causes an increase in wave drag and reduced lift due to decreased pressure under the wing. The airflow may exceed Mach 1.0 at the point of maximum pylon thickness, which is also the channel area's least cross-section distance. The purpose of the Kutney bump is to modify the airflow through the channel and to avert the effects of supersonic flow and prevent a shockwave from forming. An alternate method is to use an engine pylon with the inboard side curving at a greater radius.

The American, John Kutney of Ohio, USA, designed the Kutney bump circa 1979.

*Compare with* **Compression pylon.**

**Kutta condition** The Kutta condition, discovered by Professor Wilhelm Kutta (1867–1944) is the relationship between the smooth airflow circulation around the wing and the resulting lift it produces.

During his aerodynamic research in 1902, Professor Wilhelm Kutta noted the airflow, after passing over and under the wing departed the wing's trailing edge as a smooth flow with a downward component resulting in circulation and correctly concluded that lift

was associated with circulation. This discovery became known as the Kutta condition, its importance being used by aerodynamicists in theoretical analysis when calculating the lift of a wing. The Kutta condition exists when the airflow is a steady circulation around the wing with a stagnation point developed on the wing's lower leading edge and a smooth flow off the trailing edge.

The conditions at the trailing edge are the important feature of the Kutta condition. If the trailing edge has an angle between the upper and lower surfaces (it usually has) then the air flowing off the trailing edge will be zero at the rear stagnation point on the wing, with the airflow rejoining aft of the rear stagnation point. If a cusp is built into the lower trailing edge, then the stagnation point airflow velocity and direction will both be the same on the upper and lower surface. The airflow must also be a smooth laminar flow, not turbulent, for the Kutta condition to exist.

In 1902, Wilhelm Kutta officially published his findings on circulation with lift, as the Kutta condition. He was working along the same track as Lanchester did a few years earlier but Kutta was able to explain his findings in a more understandable way.

It is also known as the trailing edge condition.

**See Bound vortex; Cusp; Kutta-Joukowski theorem; Lifting line theory; Helmholtz vortex theorem.**

**Kutta, M.** The flights of Otto Lilienthal during the 1890s in Berlin aroused an interest in aerodynamics of Professor Martin Wilhelm Kutta (1867 – 1944) who was born in Pitschen, Germany. At the University of Munich, he was involved in aerodynamic research, contributing to the improvements of wing performance during WWII. In 1902, Martin Kutta officially published his findings on the 'circulation with lift' as the Kutta condition. He later became a professor in 1911 at the Technische Hochschule in Stuttgart, where he worked until his retirement in 1935.

**See Kutta condition.**

**Kutta-Joukowski theorem** In 1906, the Russian, Nikolai Joukowski (1847–1921) introduced his theorem, to calculate the wing's lift by using the potential flow theory. It describes how the wing's lift per unit span is directly proportional to the velocity of the wing's two dimensional, irrotational, inviscid circulation. Aerodynamicists may use this theorem combined with the lifting line and thin airfoil theories.

Joukowski (or Zhukovski) was unaware that Professor Martin Kutta (1867–1944) from Germany had conceived a very similar theorem in 1902. Joukowski published a paper on their findings with the result the theorem was widely accepted. In recognition of the work of both men, the theorem became known jointly as the Kutta-Joukowski theorem. The theory was the basis for understanding thin airfoil lift theory.

The Kutta-Joukowski theorem states: Lift per unit span =  $\rho V \Gamma$

Where  $\rho$  = air density

$V$  = velocity of free-stream air flow

$\Gamma$  = circulation of airflow.

See **Bound vortex; Helmholtz vortex theorem.**

The theorem is also known as the Kutta-Zhukovsky theorem, Zhukovsky being the Russian spelling for Joukowski. Some authors do not acknowledge Professor Martin Kutta's contribution to the theory, and simply refer to it as the Zhukovski theorem.



# L

**Lachmann, G. V. Dr.** Gustav Victor Lachmann (1896–1966) was a German WWI fighter pilot and an aeronautical engineer. Following a stall and crash in his aircraft in 1918, he invented the leading edge slot, but failed to gain a patent for it. He later moved to England where he gained for himself an English wife and employment with the Handley-Page Aircraft Company where he became their chief designer. However, when World War II broke out he was interned for a short time, before returning to work for Handley-Page. Handley-Page went on to develop his leading edge slot design. The slot became well known as the Handley-Page leading edge slot. However, Dr. Gustav Lachmann received little credit for his invention, although it was referred to as the



**Photo:** The RAF V-bomber, Handley-Page HP-80 Victor was designed by Dr. Lachmann. This aircraft is on show at the RAF Museum, Cosford, England.

Lachmann flap – a name that is little known these days. However, one of his greater achievements was the design of the Handley-Page HP-80 Victor bomber, which first flew in 1952. Handley-Page named the bomber after Dr. Lachmann's middle name of Victor.

*See* **Handley-Page, Sir F.; Lachmann flap.**

**Lachmann flap** The Lachmann flap is the name given to the leading edge slot invented by Gustav Victor Lachmann (1896–1966) and developed as the Handley-Page slot.

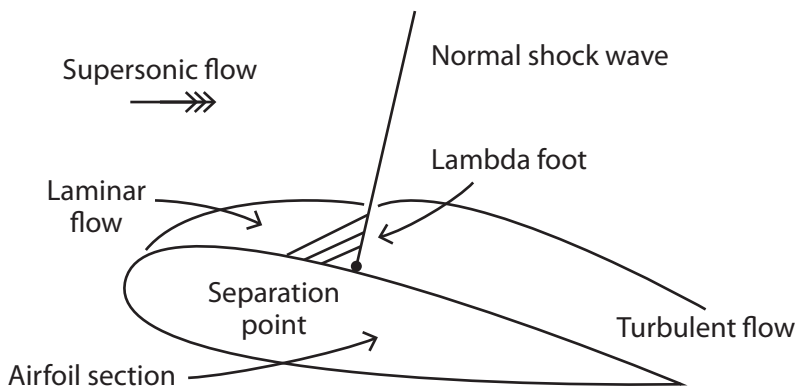
It is interesting to note Dr. Albert P. Thurston also claimed to be the inventor of the leading edge slot but after being contested in court, Lachmann's claim won the case.

**Lambda shockwave** A Lambda shockwave is named as such due to its likeness to the shape of the Greek letter Lambda ( $\lambda$ ).

The Lambda shockwave consists of a normal shockwave at approximately 70% MAC, angled slightly rearwards with a smaller oblique shockwave at its forward base, known as a Lambda foot (also known as a compression corner). It forms on a surface with a subsonic, laminar boundary layer and above the boundary layer's separation point. The effect of the normal shockwave spreads forward inducing a high-pressure area at its base on the wing surface causing a Lambda shockwave to form, tilt rearwards and to join up with the main normal shockwave. The flow becomes subsonic after passing through the shockwave and the pressure rises causing a thickening of the boundary layer to become a turbulent flow with separation from the wing's surface. This causes wave drag. A Lambda foot is common under most strong shock waves.

*Compare with* **Expansion flow & fan; Compression flow; Shockwaves.**

**Laminar boundary layer** A laminar boundary layer is found in areas of decreasing air pressure and consists of several thin laminar layers of air molecules, all sensitive to surface roughness and all following a parallel path above the surface of the wing. It is devoid of any mixing or disturbances. The layer in contact with the wing's surface may have zero velocity – any dust on the wing before flight may still be there after the flight has terminated. For each successive laminar layer, the speed increases due to the viscosity of the air – like a pack of playing cards being thrown on to a table where the uppermost cards travel the furthest distance. The top of the boundary layer is assumed the point where the velocity in the layer has reached 99% of the free stream velocity.



**Diagram 47, Lambda Shockwave**

The boundary layer is very thin at about 1.58 mm (1/16 inch) thick at the wing's leading edge and gradually thickens as it moves aft across the wing where it eventually becomes a turbulent boundary layer. In a laminar flow, the boundary layer drag (skin friction) is almost negligible, but increases as the layer becomes more turbulent. A laminar boundary layer will separate from the surface (or trip) more readily than a turbulent boundary layer.

Modern high-performance aircraft are now designed with laminar flow airfoils where a major portion of the airflow remains laminar over most of the wing.

*See **Boundary layer; Turbulent flow; Diagram 1, Adverse and Favourable Pressure Gradients.***

**Laminar drag bucket** *See Drag bucket.*

**Laminar flow** A laminar flow is the smooth, undisturbed, streamlined airflow over an aircraft's wing in flight. Air pressure within a laminar airflow will decrease with an increase of airflow speed. However, the airflow velocity within the laminar boundary layer is lower than in a turbulent layer and therefore, the flow becomes turbulent and separation will occur more readily. The chord distance for separation to occur is related inversely to the Reynolds number: at low Reynolds number, the flow is laminar.

The word laminar is from the Latin 'lamire', which means layers of laminae.

Laminar (and turbulent) flows were first studied in 1839 by the Russian G. H. L Hagen (1797–1884). In 1883, Osborne Reynolds studied the transition phase between laminar and turbulent flows.

*See **Laminar boundary layer.***

**Laminar flow airfoils** A laminar flow airfoil is designed with a maximum thickness further aft than a conventional airfoil, at around the 50% MAC location with a laminar flow covering as much of the airfoil as possible.

The symmetrical laminar airfoil is of less depth than conventional airfoils with a sharper, smaller leading edge radius. The advantage of these airfoils is the boundary layer remains attached to the upper surface and flows smoothly for a greater distance over the top surface to the transition point, which is located further rearwards than on a conventional airfoil to about 60% MAC. A 50% or more

drag reduction is also realized on laminar flow airfoils along with a delay in shock wave formation.

They are especially well suited for high-speed aircraft, due to their favourable characteristics; however, they are also found on some light aircraft. The pressure distribution over these airfoils is spread across the wing chord more evenly with a gradual increase in airflow velocity over the top surface to the point of maximum camber (about 50% MAC). At the design cruise speed, the same lift is produced as on a thicker, more cambered wing.

The disadvantage of laminar flow airfoils is their lower value of maximum lift coefficient resulting in a higher stall speed and their less than favourable stalling characteristics, combined with the rapid movement of the centre of pressure with increasing angle of attack. Any unevenness of the upper surface (ice, raindrops and round-head rivets, etc) will cause the transition point to move



**Photo:** The Consolidated B-24 Liberator with laminar flow wings, is located in the National Museum of the US Air Force in Dayton, Ohio.

rapidly forward turning the laminar flow into a turbulent flow with the associated loss of lift.

NACA Langley in Virginia developed the laminar flow wing in 1938, and it was first used on the North American P51 Mustang and the Consolidated B-24 Liberator bomber, both of WWII fame.

See **Boundary layer**; **Natural laminar flow airfoils**.

**Laminar separation bubble** See **Separation bubble**.

**Laminar sub-layer** The air flowing over the upper surface of a wing commences as a laminar flow from the leading edge and eventually breaks down to a turbulent flow as it moves further aft across the wing. However, below the turbulent boundary layer the airflow adjacent to the surface may well remain as a very thin laminar flow, known as the laminar sub-layer.

See **Boundary layer**.

**Laminar transition point** See **Transition point/region**.

**Lanchester, F. Dr.** Dr. Frederick William Lanchester (1868–1946) was a British car maker, engineer, and self-taught aerodynamicist. He was the founder in 1884, of the circulation theory of lift, which he presented to the Physical Society of London in 1897.

Lanchester went on to develop the lifting-line theory – another viewpoint on how a wing produces lift. He also made the first studies on wingtip vortices. Lanchester's style of writing in 1907 and his practice of introducing his own terminology made his theories rather difficult for others to understand and accept. However, his detailed writings on aerodynamics published as two volumes of *Aerial Flight* in 1905 and 1907 were the first to be published since Sir George Cayley's (1773–1857) were published circa 1809–10. Lanchester's first volume titled *Aerodynamics*

presenting his circulation theory of lift was the foundation for modern aerodynamics, as we know it today.

Joukowski and Martin Kutta further expanded the circulation theory of lift and the lifting line theory. From 1909 to 1920 Dr. Lanchester was a member of the Advisory Committee of Aeronautics for the British government.

*See* **Circulation theory of lift ( $\Gamma$ ); Lifting line theory.**

**Lanchester-Prandtl theory** *See* **Lifting line theory.**

**Langley Research Centre** The Langley Research Centre was founded on 17 August 1917 to become the first aeronautical laboratory in the USA. It has received several name changes over the years. It was originally known as the NACA Aviation Experimental Station and Proving Grounds, which later became Langley Field, Hampton, Va, USA. The name was changed to the Langley Aeronautical Laboratory, which in turn became the Langley Memorial Aeronautical Laboratory. Today it is known as the Langley Research Centre.

Many famous names in aeronautical research worked here, men such as Eastman Jacobs, John Stack, Theodore Theodorsen, Max Munk, Frederick Weick and after WW II, Dr. Adolph Busemann, Richard Whitcomb and many more.

Several wind tunnels were built and used for aeronautical research over many years, as shown in chronological order below:

- 1920. One of the first tunnels to be built was the five-foot open circuit type wind tunnel operational from 11 June 1920. It was used for research into speeds of up to 120 MPH.
- 1923. A variable density wind tunnel, proposed by Max Munk from Gottingen, became the primary source for aerodynamic data at high Reynolds number and the testing of a variety of airfoils resulting in many great aircraft such as the Douglas DC-3, Boeing B17 Flying Fortress, and Lockheed P-38, for example. The famous series of NACA airfoils evolved from here.

- 1927. During the time of extensive propeller research, the wind tunnels also proved the need for retractable undercarriage to reduce drag.
- 1928. NACA's first high-speed wind tunnel at Langley had a 1-foot diameter test section. Eastman Jacobs and John Stack (head of Langley Memorial Aeronautical Laboratory Compressibility Division at that time) discovered shockwaves on models when they used an early form of the schlieren optical system. Fred Weick developed the NACA engine cowl here in 1928, which greatly improved engine cooling and reduced the drag on radial engines. The Lockheed Vega was one of the first aircraft to benefit from the NACA cowl.
- 1930s. In the mid 1930s, an eight-foot high-speed wind tunnel was built at Langley capable of operating with speeds up to 500 MPH. The world's high-speed aerodynamic research was led by NACA in the late 1930s and still continues today under the guidance of the present NASA at research centres such as the one at Langley.
- A full-scale wind tunnel was also built at Langley, which became operational in 1931. Many of the American built World War II aircraft were tested in this tunnel, which closed in 2009.
- A 20 feet vertical spin tunnel was built here in 1941.
- 1950. The National Transonic Facility at NASA Langley opened their 8-feet, slotted-throat, transonic wind tunnel. The tunnel enabled large size models to be tested under various atmospheric conditions at transonic Mach numbers. Considerable new data was obtained for the development of high-speed aircraft.
- During 1950–60s, four hypersonic wind tunnels were built at NASA's Langley Aerothermodynamics Laboratory for the study of hypersonic aerodynamics. The tunnels are capable of research in the range of Mach 6 to 18. Computational aerothermodynamics is used in the study of aerodynamic performance, fundamental-flow physics research and advanced hypersonic aerospace-vehicles for the future.



**Langley, S.** Samuel Piermont Langley (1834–1906) a physicist, astronomer, and aviation pioneer. He was born in Roxbury, Mass, USA. In the late 1880s, Langley developed an interest in aerodynamics and had built two model aircraft by 1896. He designed and built a full-scale plane piloted by Charles Manley (not Langley himself) that attempted to fly on 7 October 1903; The flight was unsuccessful and ditched in the Potomac River. From 1887–96 he studied the effects of the movement of the centre of pressure on a flat plate airfoil and discovered it must be located forward of the centre of gravity. He also noted the centre of pressure moves rearward on a flat plate with an increase in angle of attack; the opposite occurs on a cambered wing where it moves forward.

Professor Ludwig Prandtl (1880–1953) explained the movement of the centre of pressure for a cambered airfoil during World War 1.

**Laplace's equation** Pierre-Simon Laplace (1749–1827) was a French mathematician and astronomer. He developed the now well-known equation, which is used in mathematical calculations in physics and aerodynamics to calculate an incompressible, irrotational flow. Laplace's second-order partial differential equation is beyond the scope of this book.

**Large area flaps** *See* **Barn door flaps.**

**Lateral axis** The OY lateral or transverse axis is a straight line at  $90^\circ$  to the OX longitudinal axis acting through the centre of gravity. The aircraft rotates nose-up or down about the lateral axis.

It must be clarified here, when the aircraft's nose pitches up, it is considered a positive movement with respect to the OY lateral axis, albeit de-stabilizing. Conversely, pitching down is a negative movement but stabilizing.

The lateral axis is also known as the pitch axis.

*See* **Axes.**

**Lateral dynamics** Roll subsidence mode, Dutch-roll mode and the spiral mode all describe the aircraft's lateral dynamic motion.

*See Dutch roll; Roll subsidence mode; Spiral mode stability.*

**Lateral dynamic stability** An airplane's lateral dynamic stability combines the effects of lateral and directional stability, which can produce directional and/or spiral divergence or Dutch roll. These are listed as follows:

- the airplane's moment of inertia about its yaw and roll axes
- the sideslip aerodynamic forces
- the rolling moment due to yaw and rolling velocity, lateral stability, and roll damping
- the yaw moment due to static directional stability, rolling, or yaw velocity, yaw damping or adverse and proverse yawing.

*See Divergence; Spiral divergence; Dutch roll.*

**Lateral oscillation** A periodic motion involving roll, yaw, and sideslip is a lateral oscillation. It is also known as **Dutch roll**.

**Lateral quality** The lateral quality refers to how well the aircraft behaves in the rolling plane.

**Lateral static stability** An aircraft's lateral static stability is determined by its reaction to a displacement from straight and level flight around the OX longitudinal axis. An upset will cause a roll followed by a sideslip and a righting moment if the aircraft is laterally stable.

A sweptback wing induces lateral static stability when the aircraft yaws/rolls to either side in turbulence. Consider a yaw to the right. Due to inertia, the aircraft will continue moving in the same direction even with the nose pointed to the right. The left wing will experience an effective greater span with a reduced effective span on the right wing. Because the fuselage shields part of the right wing near the root, its effective lifting area is reduced to an amount less than the left wing. Less area means less lift. Therefore, the left wing produces more lift and drag than the right wing and will create a restoring moment to reduce the yaw and level the wings. In addition, due to the oncoming airflow passing over the left wing at a different angle than the right wing the effective thickness/chord ratio is increased, again producing greater lift. Conversely, the right wing will present a lesser effective thickness/chord ratio producing less lift and less drag. The imbalance of lift and drag on each wing tends to restore the aircraft back to straight and level flight.

An aircraft with a high wing and low centre of gravity will produce a righting couple to restore the aircraft back to straight and level flight (pendulum stability) as also will a low wing with dihedral. The degree or strength of the inherent lateral stability determines the outcome of the restoring moment.

Up to the first few degrees angle of bank (depending on aircraft type) the aircraft has a well-damped aperiodic motion, which will naturally tend to return the aircraft to a wings level attitude; the aircraft has roll subsidence and is therefore laterally stable. Beyond a certain angle of bank, the wings will tend to diverge further into the roll, due to the outer wing travelling faster than the inner wing; the aircraft has a reduced damped aperiodic motion. If left uncorrected by the pilot, the aircraft will enter a spiral dive (the

graveyard spiral). Remember roll causes yaw and yaw causes roll, both movements being interrelated.

The degree of lateral static stability can be calculated (where altitude, speed, and weight are ignored) from the rolling moment coefficient formula:

$$\text{Roll moment } L = C_l q S b$$

Where  $L$  = rolling moment, ft-lbs

$C_l$  = rolling moment coefficient

$q$  = dynamic pressure, psf

$S$  = wing area, sq.ft.

$b$  = wing span, ft.

G. H. Bryan completed the first mathematical study of an aircraft's lateral motion in 1916.

*See Dihedral action/effect; Dutch roll; Rolling damping moment; Stability.*

**Law of continuity** *See Continuity equation.*

**Leading edge** The wing's leading edge is, obviously, the front part of the wing. This is a rather loose term as no chordwise measurement is defined. The term leading edge is applied to the main wings, tail surfaces, and rotor or propeller blades.

**Leading edge cuff** A leading edge cuff is usually found on light aircraft wings to maintain the airflow over the wing's upper surface at high angles of attack. This ensures a lower stalling and approach speed, a requirement for shorter landings. Aileron control will be maintained or improved due to the airflow staying attached over the wing and also, the stall itself will be more docile compared to a wing with a smaller radius leading edge.

It is also known as a droop.

**Leading edge devices (LEX)** Slat, slots, dogtooth, Krueger, and leading edge flaps are all encompassed under the heading of leading edge devices.

They alter the profile of the leading edge by changing a sharp, small radius edge into a blunt edge of larger radius. This has the advantage of reducing the leading edge peak pressure, re-energizing the boundary layer across the wing and through the adverse pressure gradient for an overall increase in maximum lift coefficient. By increasing the lift over the forward part of the wing at relatively high angles of attack, it improves the wing's stall characteristics while lowering the stall speed.



**Photo:** The Scottish Aviation Pioneer CC1 achieves STOL performance with the help of its leading edge flaps. This plane is on view at RAF Museum Cosford, UK.

Leading edge devices are more effective on take-off than on landing due to the fact they increase the lift, which is required with little effect on drag. Trailing edge flaps on the other hand are more

effective on landing – they increase the required drag more than the lift. Mounted on fighter aircraft, they improve the in-flight handling in extreme attitude manoeuvres.

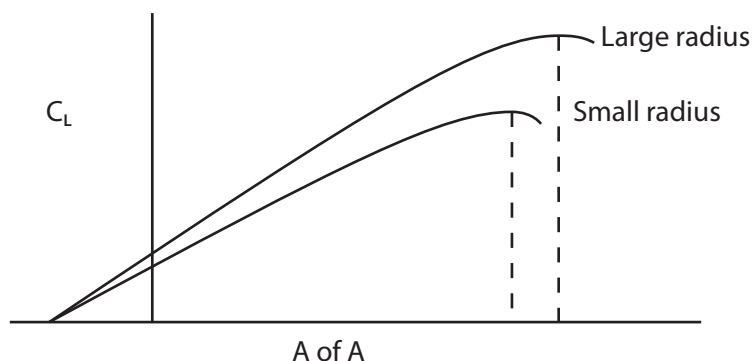
The American terminology is leading edge extensions.

*See* **Slats; Slots; Krueger flaps; Leading edge devices (LEX); Powered slat.**

**Leading edge extensions** *See* **Leading edge devices (LEX).**

**Leading edge notch** *See* **Dog tooth.**

**Leading edge radius ( $\varphi$ )** The radius is a measure in percent of the airfoil's chord and it determines how rounded (or pointed) the wing's leading edge is. A zero radius of 1–2% is like a knife-edge as found for example on the Lockheed F104 Starfighter. A wing with a small radius leading edge will have a lower lift coefficient and adverse stall characteristics as opposed to wings designed for low-speed and high lift, which have well-rounded, large radius leading edges, docile stall behavior and produce a high lift coefficient.



**Diagram 48, Leading Edge Radius**

Large radius leading edges are unsuitable for supersonic aircraft because they produce stronger shockwaves, caused by the airflow around the leading edge having to follow a greater curvature and change in direction. The change in direction is less on small leading edge radius wings, which reduces the chance of a bow shockwave from forming and creating excessive drag. High-speed airfoils with a small leading edge radius use leading edge devices to increase the radius for improved low speed handling, as on take-off and landing.

The leading edge radius can be described as a circle connecting the leading edge camber with points tangent to the upper and lower surface of the airfoil.

*See* **Leading edge devices (LEX).**

**Leading edge root extension (LERX)** Chines and strakes fall under the heading of LERX.

Root extensions are required to produce vortex lift at high angles of attack on swept wing and delta wing aircraft. A leading edge root extension is part of the wing root, extending forward of the wing's main leading edge. It can be in the form of an extended chine or strake as found on some fighter aircraft and on Concorde. On light aircraft such as the Piper PA-28 Cherokee the root extension is there not so much for aerodynamic reasons (although it does increase the wing root area) but to streamline the airflow over the wing's forward attachment point.

*See* **Forebody strake; Chine.**

**Leading edge slots** *See* **Slots.**

**Leading edge suction** The wing's entire upper surface is an area of decreased air pressure lower than the ambient pressure, but the peak decrease in pressure is greatest near the leading edge; it varies with the ratio of lift to air speed. The leading edge suction accounts for automatic slats being pulled out from the wing's leading edge with increasing angle of attack assisted on some aircraft, by springs. The leading edge suction forms above and below the leading edge stagnation point, which is a small area of increased stagnation pressure.

*See Diagram 65, Pressure Differential & Stagnation Points; Slats; Polhamus suction analogy.*

**Leading edge sweep** The sweepback angle of the wing's leading edge relative to the aircraft's longitudinal axis, is usually greater than the trailing edge sweep angle.

**Leading edge vortex** *See Vortex lift.*

**Lead-lag** Lead-lag, also known as dragging, allows freedom of angular movement of the helicopter's rotor blades in the fore and aft plane of rotation. Faulty drag dampers or rotor blades out of balance can cause the rotor head mast to wobble and incur violent and destructive oscillations.

Drag hinges, which allow lead-lag to occur and prevent the above problems, are found in rotor systems with three or more blades; two-blade rotor systems are not affected by lead-lag problems.

*See Drag hinge; Ground resonance.*

**Left-handed propeller** A propeller that rotates top blade down turning to the left, or anti-clockwise when viewed from the rear of the propeller, is termed a left-handed propeller.

*See Handed propellers; Right-handed propeller; Asymmetric thrust.*



**LEMAC** An abbreviation for 'leading edge mean aerodynamic chord'. Pilots of large aircraft use this term for centre of gravity calculations. The C of G is calculated as distance measured from the leading edge as a percentage of the wing's mean aerodynamic chord.

*See* **Chord; Mean aerodynamic chord (MAC).**

**Lewis, Dr. G. W.** Dr. George William Lewis (1882–1948) was the Director of Aeronautical Research, at NACA HQ in Washington, USA, until 1947.

**Lewis Research Centre** The Lewis Research Centre in Cleveland, Ohio was established in 1941 by NACA (later NASA) as a research centre for aircraft propulsion. It was named in honor of Dr. George Lewis and in 1999, it was renamed after the astronaut and politician as the John H. Glen Research Centre.

**Lift** The lift force is perpendicular to the free stream airflow direction and is represented by the vertical component of the resultant aerodynamic reaction force acting on the aircraft's wing in flight.

Lift is asymmetric being greater on the upper surface and less on the lower surface of the main wings and the reverse generally, but not always, on the horizontal tail. Therefore, lift is caused by the pressure differential between the upper and lower airfoil surfaces. The increase in velocity of the airflow over the upper surface will rejoin the lower surface flow aft of the trailing edge. However, due to the difference in velocity of a given parcel of air, which is divided into two separate flows at the leading edge, has the faster upper flow reaching the trailing edge earlier than the lower flow, despite the greater distance covered over the wing's curved, upper surface.

In normal straight and level flight, lift is considered to be equal and opposite to the aircraft's weight. However, lift is greater than weight during a turn and acceleration manoeuvres but less than weight during a normal climb or descent! If a down force exists on

the tailplane, as it sometimes does, the force must be counteracted by extra lift from the main wings; the lift is then termed the net lift.

The  $V^2$  in the lift formula when applied to helicopters refers to the speed of the rotor blades – not the speed of the helicopter itself.

The lift produced by the wing depends on several factors:

- air density
- angle of attack
- circulation
- compressibility effects
- flow velocity squared
- Reynolds number (air viscosity or drag)
- shape of airfoil section
- wash-in or wash-out (wing twist)
- wing condition (ice formation, rivets and general smoothness)
- wing planform (aspect ratio, area, straight or swept back and taper, etc).

Four of these factors appear in the lift equation formula:

$$\text{Lift} = C_L \frac{1}{2} \rho V^2 S$$

Where  $C_L$  = lift coefficient

$\frac{1}{2}$  = a constant

$\rho$  = air density  $\text{kg/m}^3$  or  $\text{lbs/cu.ft.}$

$V^2$  = aircraft (or rotor blade) speed in  $\text{m/sec}^2$  or  $\text{FPS}^2$

$S$  = wing area in square meters or square feet.

Various theories have been introduced to describe how lift is produced; there are advantages and disadvantages in each theory. No one theory is strictly correct in its own right, rather the different theories should all be considered. They include the following:

- |                                                           |                              |
|-----------------------------------------------------------|------------------------------|
| • <b>Bernoulli's theorem</b>                              | • <b>Path length</b>         |
| • <b>Circulation theory of lift (<math>\Gamma</math>)</b> | • <b>Thin airfoil theory</b> |
| • <b>Downwash</b>                                         | • <b>Vortex lift.</b>        |
| • <b>Kutta condition</b>                                  |                              |
| • <b>Lifting line theory</b>                              |                              |
| • <b>Newtonian theory of lift</b>                         |                              |

*See the individual entries for the lift theories listed above.*

**Lift augmentation devices** Leading edge slats and slots, droops, Krueger flaps and trailing edge flaps are all included under the heading of lift augmentation devices. *Diagram 49, Lift Coefficient V. Angle of Attack* shows the increase in maximum lift coefficient with flap and leading edge devices deployed. When the flaps are lowered the centre of pressure moves aft pitching the leading edge nose-down. The tailplane download may change due to changes in the downwash off the main wing.

See **Flaps; Leading edge devices (LEX).**

**Lift boost** See **Form thrust; Vortex lift.**

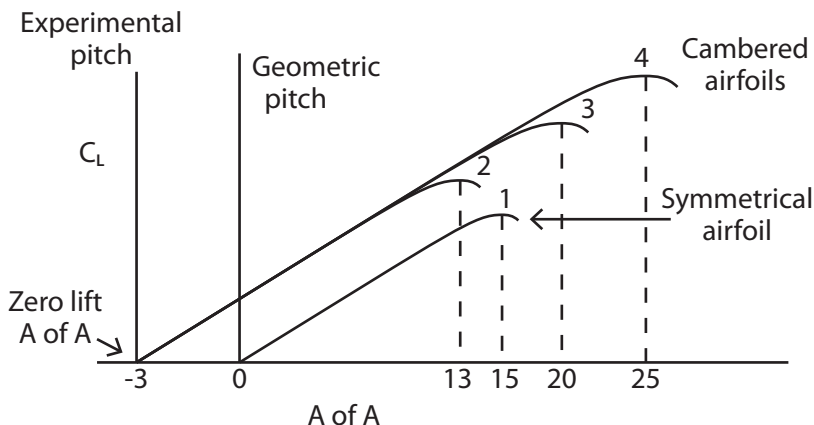
**Lift coefficient** The lift coefficient is a non-dimensional number relating the quantity of dynamic pressure and wing area converted into lift; it is a function of the angle of attack, Mach and Reynolds numbers and aircraft configuration. It is also a measure of how efficient the wing is in producing lift. The higher the lift coefficient, the greater the lift that can be produced by the wing. The lift coefficient can be shown by the formula:

$$C_L = \frac{\text{Lift pressure}}{\text{dynamic pressure}} = \frac{L}{qS}$$

The lift coefficient increases in proportion to the angle of attack, which increases linearly by approximately 0.1% per degree angle of attack. This is shown plotted on a graph as a straight-line slope rising up to the peak of the maximum lift coefficient, where the wing produces its maximum lift; any further increase in angle of attack results in a reduction in lift coefficient, a loss of lift and an aerodynamic stall. Each lift coefficient curve is initially of similar slope angles. The lift coefficient depends on:

- the airfoil section's camber including trailing edge flaps
- the leading edge radius including leading edge slats or slots
- the angle of attack, which is changed by pilot input

- the condition of the boundary layer, icing, and surface smoothness, etc.



**Diagram 49, Lift Coefficient V. Angle of Attack**

The maximum lift coefficient increases with increased wing area and camber, but for a sweptback wing, it is less than any equivalent airfoil on a straight wing. This is reflected in the lift coefficient curve for the sweptback wing being placed below the straight wing's curve as shown on **Diagram 52, Lift Coefficient V. Straight, Swept and Delta Wings**. The Reynolds number does not affect the lift coefficient slope; however, the maximum lift coefficient is affected by the Reynolds number due to the viscous forces in the airflow. The maximum lift coefficient of a wing determines the stalling speed, where a greater lift coefficient lowers the stalling speed. The thin airfoil theory or lifting line theory can be used to find the approximate lift coefficient. Reference to **Diagram 49, Lift Coefficient V. Angle of Attack**, shows the lift coefficient for a symmetrical airfoil and cambered airfoil with flaps up and down, and with leading edge slots extended (listed 1 to 4 on the diagram). It also shows that the wing or propeller's experimental pitch produces zero net thrust at approximately minus two degrees

angle of attack and the geometric pitch ceases to produce thrust at zero degrees angle of attack.

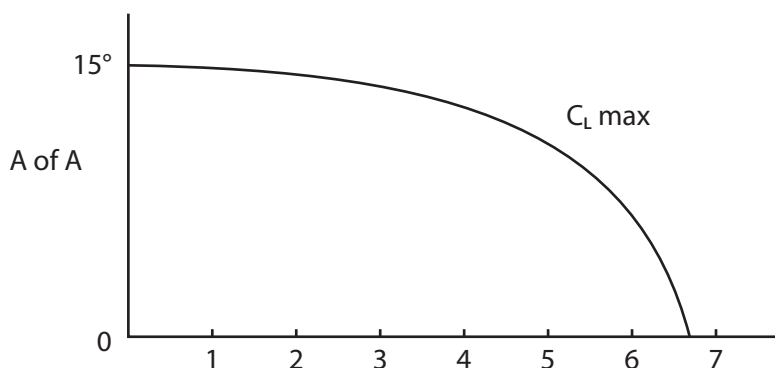
1. Symmetrical airfoil – no flaps
2. Cambered airfoil – plain flaps up
3. Cambered airfoil – plain flaps down
4. Cambered airfoil – leading edge slots and Fowler flaps extended.

The lift coefficient is also referred to as the aircraft lift coefficient, or planform coefficient. The lift coefficient curve is also known as the lift curve, which denotes the lift coefficient only, not the total wing lift.

**See Angle of attack ( $\alpha$ ); Lifting line theory; Critical angle of attack; Thin airfoil theory; Zero lift.**

**Lift coefficient & angle of attack** Depending on the airfoil's camber and sharpness of the leading edge radius, the maximum lift coefficient occurs between about  $12^\circ$  and  $20^\circ$  angle of attack and at a relatively low air speed. With increasing air speed, and thus decreasing angle of attack, the boundary layer changes from an attached laminar flow to a detached turbulent flow and eventually breaks down and separates from the airfoil's surface at the separation point. At high angles of attack, and reducing air speed, the airflow has difficulty in following the upper contours of the airfoil, due to the high adverse pressure gradient. As the speed decreases, the airflow breaks away more readily, the airfoil stalls and the angle of attack reduces. It follows, the boundary layer is a laminar flow only within a given range and at each end of the angle of attack range, a turbulent flow develops on the rear portion of the wing. The reference lift coefficient occurs at zero degrees angle of attack.

Reference to **Diagram 50, Maximum Lift Coefficient V. Air Speed**, shows the maximum lift coefficient reduces with increasing air speed at a constant angle of attack. This corresponds to the lift



**Diagram 50, Maximum Lift Coefficient V. Air Speed**

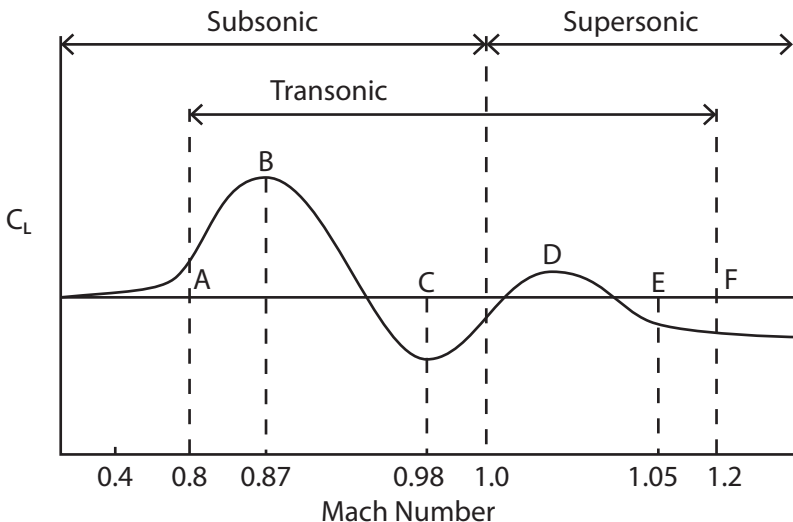
slope curve decreasing with reduced angle of attack (**Diagram 49, Lift Coefficient V. Angle of Attack**). NACA engineer Lyman J. Briggs discovered this fact in the 1920s and also the following:

- the drag coefficient increases with increasing speed
- the centre of pressure moves rearward with increasing speed
- the critical Mach number decreases with increasing angle of attack and airfoil thickness.

It accounts for the retreating blade stall on a helicopter's main rotor blade where the blade tip stalls first and the stalled area spreads inboard a short distance, even though the blade tip is the fastest moving part of the blade. The blade root remains unaffected even though the angle of attack at the root is greater than at the tip and it is moving a lot slower. The blade tip becomes affected due to the maximum lift coefficient decreasing with increasing speed, as explained above.

**Lift coefficient & compressibility** For a given angle of attack, the lift coefficient is constant in low-speed subsonic flow but varies substantially in transonic and supersonic flow due to the effects of compressibility.

*Diagram 51, Lift Coefficient & Mach Number*, shows the variation of the lift coefficient from about Mach 0.3 up to the supersonic range at Mach 1.2, assuming a subsonic, constant thickness/chord ratio wing, at a constant angle of attack. Initially the lift coefficient line remains horizontal but starts to rise around 200 KTAS/Mach 0.3 when compressibility starts to manifest itself. The airflow over the wing starts to change its pattern as the streamlines above the wing merge and the upwash at the leading edge rises to meet the wing at a more acute angle. These two effects combine to increase the lift coefficient shown by its upward curvature towards the peak on the diagram. Position A on the curve marks the position of the critical Mach number where the first of the shockwaves start to form on the wing's upper surface.



**Diagram 51, Lift Coefficient & Mach Number**

The lift coefficient peaks at position B on **Diagram 51, Lift Coefficient & Mach Number**, where the shockwave's intensity increases causing the pressure gradient to become more adverse with an accompanying thickening of the boundary layer and separation of the flow. The separation point moves forward on the wing, the lift (and lift coefficient) decrease resulting in a shock stall at approximately Mach 0.87 or so.

At position C on **Diagram 51, Lift Coefficient & Mach Number**, the lift coefficient has reduced to its lowest limit. The upper surface shockwave continues to strengthen and moves aft to around 70% MAC. It is then accompanied by the formation of a shockwave on the wing's under surface; they both move to attach to the wing's trailing edge and are then known as a fishtail shockwave. The whole wing is then immersed in a supersonic flow with a high-pressure region on the rear of each surface – the wing becomes rear loaded with a loss in lift. The lift coefficient curve plummeting to its lowest value at position C and a Mach number of approximately Mach 0.98 shows this loss. With the shockwaves stabilized, the wake is reduced with an improvement in lift again and the lift coefficient curve once again rises to a value slightly higher than found in the subsonic flow at location D.

As the Mach number increases more so, the bow shockwave forms pushing ahead of the leading edge and saps energy from the flow. The lift coefficient is reduced again to a value below the subsonic value with the speed in excess of Mach 1.05 plus (the Mach detachment speed) at location E.

The aircraft continues to accelerate into the low supersonic region and all shockwaves become oblique waves. A stable flow has settled-in with the lift coefficient becoming relatively constant at a value less than at subsonic speed, shown by location F on the diagram, at the top end of the transonic Mach range.

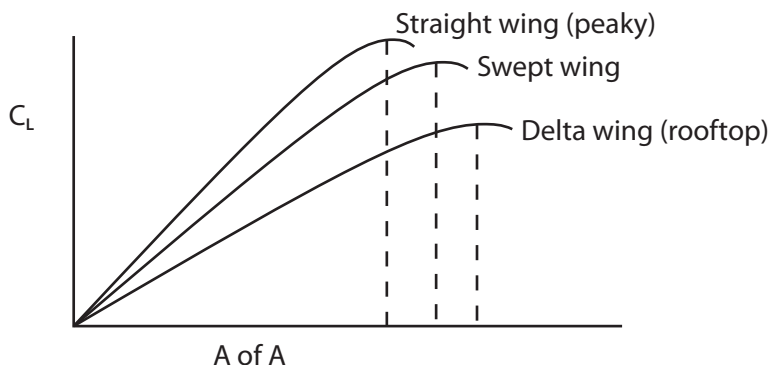
See **Compression flow; Normal shockwave; Oblique shockwave; Drag divergence curve.**



**Lift coefficient & sweptback wings** A sweptback wing produces a lower lift coefficient than a straight wing.

The critical angle of attack for a straight wing is less and the peaky curve at the higher lift coefficient shows the characteristics of a prominent stall occurring at its maximum angle of attack. In contrast, the sweptback wing produces less lift for the same angle of attack as a straight wing and a lower maximum lift coefficient, which is achieved at a higher angle of attack. The lift curve for the sweptback wing shows a more docile stall behavior than a straight wing (a rooftop curve) although the swept wing usually pitches upwards at the stall. Delta wing aircraft do not stall in the same manner as straight or sweptback wings; the stall is even more docile and the rate of climb decreasing to zero is considered as the limiting factor rather than an actual stall angle of attack. The sweptback wing rides through turbulence more smoothly than does a straight wing due to it being less affected by angle of attack changes for a comparable wing loading and aspect ratio.

See **Sweptback wings**.



**Diagram 52, Lift Coefficient V. Straight, Swept and Delta Wings**

**Lift-dependant drag** Lift-dependant drag is commonly known as induced drag and is the bane of all lifting surfaces. Due to the downwash off the wing's trailing edge, the total reaction of the wing is tilted rearwards. The horizontal component of the total reaction represents the lift-dependant induced drag. It is impossible to have lift without lift-dependant drag.

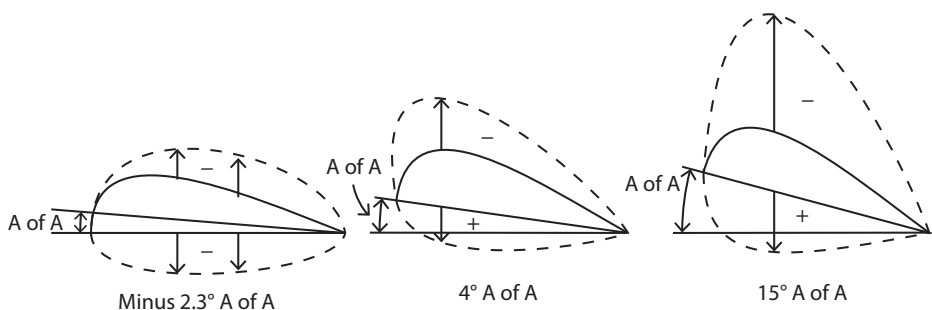
It is also known as lift-induced drag.

*See Induced drag; Total drag.*

**Lift distribution** The chordwise variation in lift on an aircraft's airfoil section and the movement of the centre of pressure varies with changing angle of attack; *Diagram 53, Lift Distribution*, shows this.

The shape of the lift's pressure pattern areas shows how the positive and negative pressure regions change with increasing angle of attack as the aircraft's speed reduces. It is the airfoil's shape, which induces the decrease in air pressure and in turn, causes the airflow speed to accelerate over the upper surface, in accordance with Bernoulli's theorem.

Reduced pressure occurs over the top of the wing and usually increased pressure underneath the wing. Under certain flight conditions, a cambered airfoil may produce some suction below the wing and thus a small downward lift force (negative lift) but to a lesser extent than the upper surface suction. However, it is the difference in pressure that provides the required lift. The downward pointing arrows on some diagrams show the suction below the wing. If the negative pressure (suction) on the top surface is the same as on the lower surface, the wing will produce zero net lift, which occurs at zero degrees angle of attack for a symmetrical airfoil and at about minus two or three degrees angle of attack for a cambered airfoil. Depending on the shape of the airfoil, either symmetrical or cambered and the angle of attack, the air pressure may vary over the wing's under surface being positive or negative on different parts of the airfoil chord.



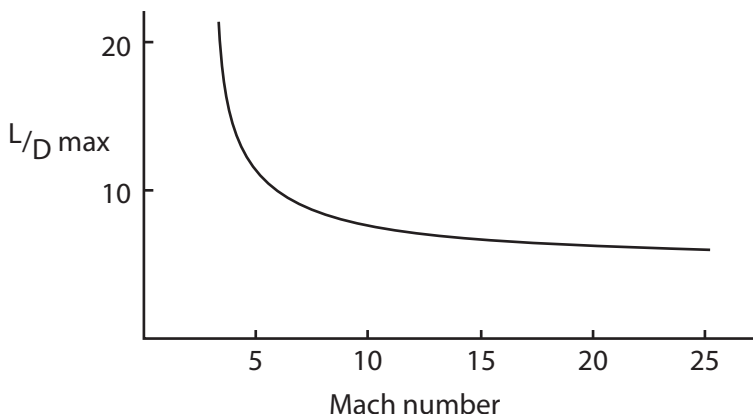
**Diagram 53, Lift Distribution**

At minus two or three degrees angle of attack on a cambered airfoil, (**Diagram 53, Lift Distribution**) the negative pressure is equal on both upper and lower surfaces, therefore no lift is produced (zero lift). Increasing the angle of attack to four degrees causes the air pressure to become negative on the upper surface and positive on the lower surface. As mentioned above, it is the difference in pressure that produces lift – the lift is asymmetric. An increase in angle of attack up to the critical angle will weaken the pressure gradient resulting in flow breakaway and a turbulent reverse flow, greatly destroying the lift over the aft portion of the wing. A small amount of lift is still produced by the lower surface of the wing due mainly to the positive pressure still being present. Beyond the critical angle, the centre of pressure moves aft.

The pressure distribution envelopes as shown in **Diagram 53, Lift Distribution**, are found by wind tunnel experiments using a manometer to measure the pressure distribution over a wing.

**Lift/drag barrier** The lift/drag barrier is a hypersonic cruise speed phenomena. **Diagram 42, Hypersonic  $C_L$ ,  $C_D$  &  $L/D$  Ratio** (the curves for hypersonic flight) shows the maximum lift/drag ratio decreases as Mach number increases. Unlike subsonic flow, high lift/drag ratios in hypersonic flight are almost impossible to achieve due to the high wave drag induced by strong shockwaves combined with high skin friction drag. Both of these factors cause a lift/drag barrier preventing relatively conventional designed aircraft from achieving hypersonic flight. To overcome this problem, waverider aircraft are specially designed to break through the lift/drag barrier and achieve higher lift/drag ratios, which enables greater range due to their increased aerodynamic efficiency.

The lift/drag barrier line on **Diagram 54, Lift/drag Barrier**, shows the approximate theoretical boundary line. The lift/drag ratio of an aircraft normally falls below the line indicating a low lift/drag ratio with increasing Mach number. The ideal would be to plot the lift/drag ratio above the barrier line, (for improved aerodynamic efficiency) which in practice is not an easy objective to achieve, hence the lift/drag barrier.



**Diagram 54, Lift/drag Barrier**

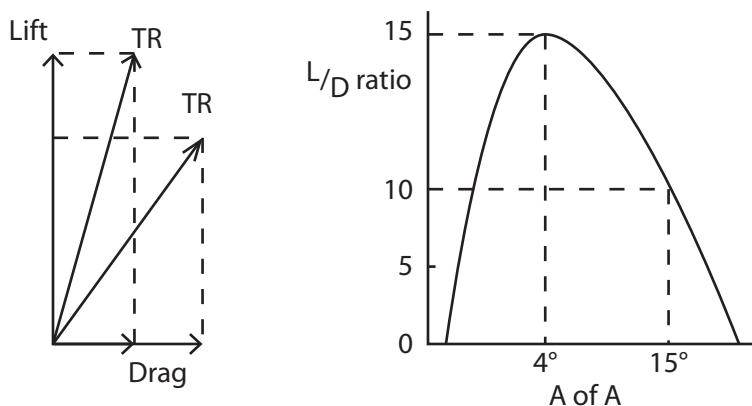
**Lift/drag ratio** The lift/drag ratio is the ratio of total lift to total drag developed by the aircraft; it is a measure of the aircraft's aerodynamic efficiency.

The most efficient angle of attack is around three or four degrees with respect to the maximum amount of lift produced for minimum drag. A high lift/drag ratio is required for greater aircraft performance and efficiency, lower fuel burn per distance gone and good weight carrying ability. The wing's maximum lift/drag ratio will be achieved at the same angle of attack at any altitude regardless of aircraft weight. The indicated air speed will also remain constant at all altitudes for a given angle of attack. However, an increase in aircraft weight will require an increase in indicated air speed to maintain a constant lift/drag ratio. The lift/drag ratio determines the aircraft's rate of climb, maximum cruising range, and descent performance. Straight wing aircraft have a lift/drag ratio of about 10:1 compared to sweptback wing aircraft where it is about 5:1. The maximum lift/drag ratio of helicopter rotor blades occurs at a given rotor RPM as set by the manufacturer.

Due to the airflow's upwash and downwash over the airfoil, the total reaction is tilted rearwards from the vertical away from the lift force component. With increased lift/drag ratio (greater lift and/or less drag), the total reaction will move closer to the vertical. Conversely, the total reaction will move away from the vertical with decreasing lift/drag ratio (less lift and/or more drag) as shown by the lift/drag ratio vector diagram on the left side of **Diagram 55, Lift/drag Ratio, Vectors & Graph**.

At any angle of attack, above or below the maximum lift/drag ratio, the aircraft's performance suffers. During a glide for example, the flattest glide or minimum rate of descent, is achieved at approximately three or four degrees angle of attack where the lift/drag ratio is the greatest. At any other angle of attack, the glide angle will be steeper, in either jet or piston-engine aircraft. The maximum lift/drag ratio is also the angle of attack (best speed) for maximum endurance and angle of climb in a jet; maximum range in piston-engine aircraft is also achieved at the maximum

lift/drag ratio. The lift/drag ratio curve is shown on the right side of **Diagram 55, Lift/drag Ratio, Vectors & Graph**.



**Diagram 55, Lift/drag Ratio, Vectors & Graph**

The lift/drag ratio compares the performance of different airfoils under similar conditions, and the change in performance of a given airfoil under different atmospheric conditions. The lift/drag ratio also applies to propeller and helicopter rotor blades. The angle between the lift and drag vectors plus the angle between the thrust and prop/rotor blade torque vectors determines the prop/rotor blade lift/drag ratio.

$$\text{The lift/drag ratio} = \frac{C_L \frac{1}{2} \rho V^2 S}{C_D \frac{1}{2} \rho V^2 S} = \frac{C_L}{C_D}$$

Where  $C_L$  = lift coefficient

$C_D$  = drag coefficient

$\frac{1}{2}$  = a constant

$\rho$  = air density kg/m<sup>3</sup> or lbs/cu.ft.

$V^2$  = aircraft speed in m/sec<sup>2</sup> or FPS<sup>2</sup>

$S$  = wing area in square meters or square feet.

Note all factors cancel out except L/D – the lift/drag ratio.

**Lift dumpers** The term lift dumpers is an alternate name for spoilers. They are flat plates mounted on the upper wing surface at approximately 60% MAC. When used in flight, they are referred to as spoilers for roll control or to reduce air speed. When in the raised position after touchdown, they are referred to as lift dumpers; their purpose is to destroy lift, placing more aircraft weight on the wheels and to produce extra drag to assist the brakes in stopping the aircraft.

See **Air brakes; Speed brakes; Spoilers; Dive brakes; Diverters.**

**Lift equation** The lift equation is used for calculations involving the wing's lift. In this equation, the angle of attack is ignored because no direct mathematical relationship exists between the angle of attack and lift. However, the lift coefficient, which does appear in the formula, depends on the airfoils shape and is proportional to the angle of attack. The total lift or lift coefficient and hence angle of attack are calculated by wind tunnel experiments or analysis. Air density, velocity and wing area are the other factors. The lift equation is as follows:

$$C_L = \frac{\text{Lift}}{\frac{1}{2} \rho V^2 S}$$

Where  $C_L$  = lift coefficient

$\frac{1}{2}$  = a constant

$\rho$  = air density kg/m<sup>3</sup> or lbs/cu.ft.

$V^2$  = aircraft speed in m/sec<sup>2</sup> or FPS<sup>2</sup>

$S$  = wing area in square meters or square feet.

See **Lift; Lift coefficient.**

**Lift-grading curve** The lift-grading curve shows the transverse (wingtip to wingtip) pattern of lift above the wing; preferably, it should be a semi-ellipse when the flaps are in the raised position. With flaps lowered, the centre of lift of each main wing is moved closer to the aircraft's centre line reducing the rolling moment and lateral stability. The elliptical lift grading curve will then have an upward bulge over the centre of the curve showing the increased lift in that area.

Due to a lower speed being used on the approach, more aileron deflection will be required in gusty wind conditions to maintain lateral control. The rate of roll will be faster and a stall could occur due to the large angle of attack being used at the lower speed. Some light aircraft are more laterally stable in gusty wind conditions with flaps raised – the centre of lift of each wing remains further outboard.

See **Elliptical lift distribution**.

**Lifting bodies** Lifting body aircraft have appeared on the aviation scene in two different eras and for two different reasons.

In 1921, Vincent Justus Burnelli (1895–1964), an aircraft designer and pioneer aviator, patented and built several biplane aircraft with airfoil shaped fuselages to act as a lifting body. His intentions were to increase the aircraft's lift by producing an airfoil shaped lifting body and hence, increase its payload ability. He improved on his lifting body theory up until the time he died in 1964.

In the 1960s, NASA also used lifting body aircraft but of an entirely different shape to Burnelli's craft. The use of a lifting body aircraft was first conceived and designed at the Ames Aeronautical Laboratory under the direction of H. Julian Allen and Alfred Eggers. They were built to research the manoeuvrability and viability of an aerodynamically designed space vehicle, for re-entry after an orbital space flight. In effect, they were the ancestors of the Rockwell International Space Shuttle, itself being a lifting body





**Photo:** The Martin-Mariette X-24B lifting body is located at the National Museum of the US Air Force in Dayton, having completed its flight test program.

aircraft. The lifting body research aircraft have very little wing area; the blended body/wing shape produced the required lift.

Three basic designs were built for testing at NASA's Dryden Flight Research Centre (Edwards Air Force Base). NASA tested their models from 1963 to 1980 as a US Military experimental program. The rocket-powered aircraft were launched from a Boeing B-52 mother ship before landing without power at Edwards Air Force Base.

The first lifting body aircraft built for NASA was the Northrop/NASA M2-F2 re-entry research vehicle. It made its first flight after being released from a Boeing B-52 mother ship on 12 July 1966 at the Edwards AFB, California. Devoid of any wings, it had a flat top,

bulging underside, raised cockpit and two fins. The Northrop HL-10 designed by NASA Langley followed the M2-F2 and made its first flight on 13 November 1968. The Northrop HL-10 was more streamlined than M2-F2 with a cambered bottom, round top, three fins, and a flush cockpit in the nose. The third research aircraft was the Martin-Mariette X-24 originally known as the SV-SP. It made its first free flight on 17 April 1969 at Edwards AFB. The X-24 'A' and 'B' models followed the X-24.

Test pilot Bruce Petersen (1933–2006) survived a spectacular crash-landing in the M2-F2 on 10 May 1967; the film of the crash was used for the opening sequence of the TV series, *The Six Million-Dollar Man*. The plane was rebuilt after the accident, modified and re-designated the M2-M3; it is now resident in the Smithsonian Museum in Washington, DC.

See **Research aircraft**.

**Lifting line theory** The lifting line theory of lift is a continuation of the circulation theory of lift, with the inclusion of the downwash and wingtip vortex effect. The lifting line is considered to run from the aircraft's centreline out to the wing tips. The lift can be calculated from the potential flow within the circulation.

The circulation theory explains how the airflow passes over and under the wing, developing a horseshoe vortex. When an aircraft starts its take-off run, a starting vortex is initially shed from the wings, which develops into a horseshoe vortex, comprising the bound and wingtip vortices. The bound vortex is named as such because it is considered to be a single vortex filament or bound vortex 'bound' to the lifting line, a fixed location on the wing.

The lifting line theory is only suitable for applying to straight, medium to high aspect ratio wings. The theory gives unsuitable test results when applied to straight, low aspect ratio wings and delta or sweptback wings.

Reference to **Diagram 40, Horseshoe Vortex**, shows only one horseshoe vortex. However, consider in reality, an infinite number

of horseshoe vortices are all located across the wingspan making up the lifting line. The bound vortex sheet flowing off the trailing edge induces a downwash. From the lifting line theory, all pilots today understand the difference in pressure above and below the wing produces lift.

Dr. Frederick William Lanchester (1868–1946) first devised the lifting line theory in 1906, but his explanation was difficult for the world to grasp. Prandtl later discovered the same theory, which he explained more clearly. Prandtl later gave credit to Lanchester as early as 1911 when the theory became known jointly as the Lanchester-Prandtl theory.

See **Bound vortex; Kutta-Joukowski theorem; Helmholtz vortex theorem.**

**Lift vector** The vertical component of the wing's aerodynamic total reaction represents the lift vector.

See *Diagram 55, Lift/drag Ratio, Vectors & Graph.*

**Lift/weight equilibrium** The four basic forces acting on an aircraft in flight are the thrust/drag and the lift/weight couples. The lift must be equal to weight to provide vertical equilibrium, where a given lift coefficient is obtained at a given angle of attack. The lift/weight equilibrium formula below shows the relationship between speed, wing loading, and lift coefficient. Note when considering the formula for true air speed (KTAS) the density ratio ( $\sigma$ ) is included; for equivalent air speed (EAS) the density ratio is omitted.

$$V = \frac{7.2\sqrt{W/S}}{C_L\sigma}$$

$$\text{or } V_E = \frac{7.2\sqrt{W/S}}{C_L}$$

Where  $V$  = KTAS

$V_E$  = EAS

$W/S$  = wing loading

$C_L$  = lift coefficient

$\sigma$  = altitude density ratio.

**Lift/weight ratio** The lift/weight ratio is the ratio of the aircraft's lift to its weight.

In straight and level flight, the load factor has a value of one when the lift equals the weight. A turning manoeuvre requires greater lift to maintain height and to provide centripetal force to cause the aircraft to turn. The lift/weight ratio is then increased to a value greater than one. For example, during a 60-degree angle of bank turn, the lift has doubled and increased the load factor to two.

**Limit & ultimate load factors** The American FAA Part 23 certification requirements for light aircraft under 12500 pounds (5670 kg) states that the aircraft structure must be able to support all loads up to the limit load factor without permanent deformation and not interfere with the safety of flight operations. The aircraft structure must also be able to support loads up to the ultimate load factor without failure for at least three seconds.

For non-aerobatic aircraft, the negative load factor is 0.4 times the positive limit and for aerobatic aircraft, the negative load factor is 0.5 times the positive limit.

The ultimate load factor is based on a safety factor of 50% (1.5) times the limit load factor. It provides for a safety margin over and above the limit load factor applicable to the category for which the aircraft is certified under the American FAA Part 23 rules.

The limit and ultimate load factor depends upon the aircraft category where:

|                    | Limit load factor | Ultimate load factor |
|--------------------|-------------------|----------------------|
| Normal category    | +3.8g and -1.52g  | +5.7g and -2.28g     |
| Utility category   | +4.4g and -1.8g   | +6.6g and -2.7g      |
| Aerobatic category | +6.0g and -3.0g   | +9.0g and -4.5g      |

Also known as the design load factor.

See **Normal category; Ultimate load factor; Diagram 39, Gust Envelope.**

**Linear acceleration** Linear acceleration is an increase or decrease of speed in a straight line, for example, during take-off or landing respectively.

*Compare with* **Angular velocity & acceleration.**

**Line of zero lift** The angle of attack where no lift is produced by the airfoil.

Reference to *Diagram 68, Propeller Pitch*, shows the line A-E is the zero lift line for a cambered airfoil. This line is also known as the zero thrust line in propeller or helicopters aerodynamics.

A line from the leading edge to the trailing edge of a symmetrical airfoil indicates the line of zero lift occurs at zero degree angle of attack.

A cambered airfoil has its line of zero lift when its angle of attack is at minus two or three degrees. Due to the curvature of the airfoil's upper surface, lift is still produced in a slightly nose-down attitude. A supersonic airfoil's line of zero lift is usually found at a positive angle of attack.

*See* **Zero lift line; Diagram 49, Lift Coefficient V. Angle of Attack.**

**Line vortex** The term line vortex is an alternative name for wingtip vortex, which is part of the wing's circulation airflow. It is associated with the lifting line theory, hence the name line vortex. It can be visible under certain atmospheric conditions.

*See* **Diagram 40, Horseshoe Vortex; Bound vortex; Cusp; Lifting line theory; Helmholtz vortex theorem; Kutta condition.**

**Lippisch, A, Prof.** The aerodynamicist, Professor Alexander Martin Lippisch (1894–1976) was born in Munich, Germany. His career began in 1920 as a designer of sailplanes, followed by over fifty aircraft designs in the following thirty years. He was appointed the Director of the Aeronautical Department of Rhon-Rossitten-

Gesellschaft. Circa 1928, he pioneered the research into tailless aircraft and delta wing aircraft. Between the two great wars, he is noted for his research work on airfoils. In particular, he developed the airfoil section theory and in 1931, he demonstrated the first powered delta wing aircraft. Initially, his delta wings were not accepted by the German aviation industry. However, in 1939, he joined the Messerschmitt aircraft company where he designed the Messerschmitt ME 163 Komet rocket-powered fighter plane with a sweptback wing. He was also instrumental in theoretical research on supersonic delta wing aircraft. In January 1947, he immigrated to America where he worked at Wright Field, Ohio, USA, where his ideas and knowledge were gratefully accepted by the American aircraft building industry.



**Photo:** The tailless Messerschmitt Me 163 Komet was designed by Professor Lippisch. This example is on show at the RAF Museum at Cosford in England.

**Load factor** The load factor (n) is the resultant of the combined forces of centrifugal force (due to angle of bank) and aircraft weight and is shown as follows:

$$n = L/W = 1/\cos \phi \text{ (phi)}$$

Where n = load factor (g)

L = lift

W = weight

$\cos \phi$  = angle of bank.

Acceleration can be produced in steep turns or when flying in turbulent conditions involving vertical gusts. In a steep turn, this can be seen as the load factor (or effective extra weight) as shown on **Diagram 91, Turning Forces**.

When an aircraft is flying straight and level, the lift equals the weight. In a turn, increasing the angle of attack is required to increase the lift force to maintain the vertical component of lift. The increased lift force is equal to the increased wing loading, which is equal to the load factor. Greater angles of bank induce higher load factors. The load factor is akin to extra weight – where extra weight raises the stall speed. The load factor for each given angle of bank is calculated from the formula above.

| Angle of Bank | Load Factor |
|---------------|-------------|
| 30°           | 1.15        |
| 45°           | 1.4         |
| 60°           | 2.0         |
| 70°           | 2.94        |
| 73°           | 3.8         |
| 80°           | 6.0         |

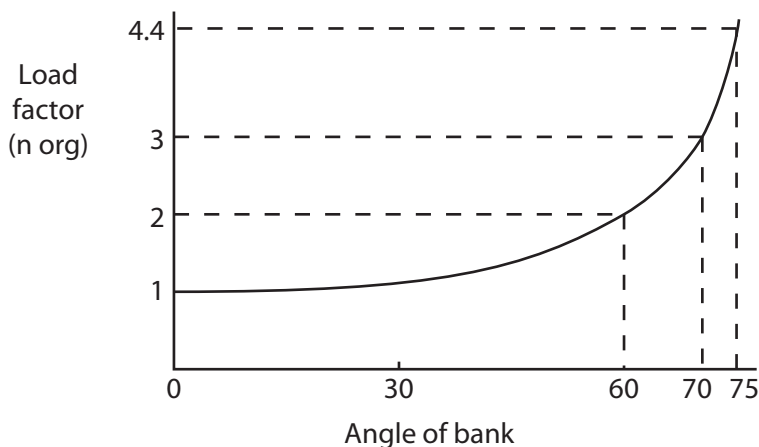
From the above table, it can be seen the load factor (determined solely by the angle of bank) increases dramatically above 60 degrees

angle of bank. **Diagram 56, Load Factor v. Angle of Bank**, shows the load factor increases exponentially with increasing angle of bank. Note, the aircraft's speed does not enter into the equation.

The American FAA Part 23 certification requirements states that an aircraft in the normal category can perform all normal manoeuvres, stalls, steep turns up to 60 degrees angle of bank, lazy eights and chandelles. The certification requirements also call for a 50% safety factor to be added to the limit load factor to record the ultimate load factor.

An aircraft can be certified in more than one category, usually with weight restrictions for operations in other than the normal category. Aircraft in the utility category can operate as in the normal category and perform spins and steep turns at an angle of bank greater than 60 degrees. The aerobatic category does not have any restrictions on flight manoeuvres except those shown necessary by the flight certification testing, for a given aircraft type.

See **Stall speed**.



**Diagram 56, Load Factor v. Angle of Bank**



**Loading** The term loading is used when referring to the aircraft's wing loading (all up weight /wing area) and the power loading (all up weight/horsepower). Loading in a turn is due to the resultant of the aircraft's weight and the centrifugal force.

**Local velocity** The local velocity is the airflow velocity over a localized area of the aircraft.

*Compare with* **Free-stream velocity  $V_\infty$ ; Relative airflow (RAF).**

**Locked-in stall** A locked-in stall is the bane of all T-tail jet transport aircraft. During a stall condition, the T-tail lies in the wake of the turbulent air flowing off the stalled main wing rendering the tail ineffective. Stick shakers and/or stick pushers are required on many jet transport aircraft to prevent or recover from a stall condition.

*See* **Deep stall; Super-stall.**

**Longer path length theory** *See* **Equal transit-time fallacy.**

**Longitudinal axis** The OX longitudinal axis is considered a straight line from the aircraft's nose to the tail passing through the Centre of gravity, usually lying on an upslope from nose to tail due to the location of the vertical fin. The aircraft rolls about the OX longitudinal axis and is therefore known as the roll axis.

*See* **Axes; Aerodynamic axes.**

**Longitudinal dihedral** To obtain longitudinal dihedral and controllability requires the tailplane to be set at an angle of incidence less than that of the main wings. Its purpose is to maintain longitudinal stability around the OY lateral axis by raising or lowering the aircraft's nose following an in-flight disturbance. The nature of the phugoid-mode oscillation is determined by the longitudinal dihedral. In other words, longitudinal dihedral is the

difference in the angle of incidence of the chord of the main wings and tailplane. However, using the chord's direction for longitudinal dihedral studies is not the most convenient choice; the zero lift axis of both main wings and tailplane is preferable, because the direction of the zero lift axis is also associated with longitudinal stability and trim, making for easier calculations.

*See* **Decalage; Longitudinal static stability; Tailplane.**

**Longitudinal static stability** Longitudinal static stability is independent of lateral and directional stability. Stability is a requirement for all aircraft in order to maintain balanced flight. A positive longitudinal stability restoring moment returns the aircraft to its original attitude after a disturbance in the pitching plane around the OY lateral axis.

The condition required for longitudinal stability is a restoring moment provided by the tailplane. The degree of longitudinal stability depends on the:

- the longitudinal dihedral
- position of the aircraft's centre of gravity and leverage arm to the tailplane
- the mainplanes and fuselage pitching moments
- tailplane area
- Angle of incidence and aspect ratio of the tailplane.

Consider the case when a wind gust causes the aircraft's nose to rise. If the restoring moment produced by an increase in the tailplane lift multiplied by the distance to the centre of gravity, is greater than the upsetting movement caused by the increase in mainplane lift and its new distance from the centre of gravity, then the aircraft will be stable. The forward location of the centre of gravity and the area and leverage of the tailplane will probably have the greatest influence. In addition, the lift on the tailplane is increased in greater proportion than the lift on the mainplanes,

causing the tail to lift. However, the movement of the centre of pressure must also be taken into account.

An aircraft in supersonic flight requires large, powerful controls to overcome the increased longitudinal static stability.

*See* **Longitudinal dihedral; Tail volume coefficient (V); Tailplane.**

**Low air density effects** The effects of low air density associated with an increase in temperature within a hypersonic flow induces the boundary layer to increase in depth and merge with the oblique chock wave.

*See* **Hypersonic aerodynamics.**

**Low drag airfoils** A wing designed to have a large area of laminar boundary layer over its upper surface with less drag from the turbulent boundary layer, is termed a low drag airfoil.

*See* **Laminar boundary layer; Laminar flow airfoils.**

**Low-speed aerodynamics** This is a loose term for the study of aerodynamics in airflows un-affected by compressibility at a speed generally accepted as below 200 KTAS/Mach 0.3. Most aerodynamic research prior to the 1940s was concerned with incompressible flow.

**Low-speed stall** A stall entered from straight and level flight with a gradual reduction in speed (usually 1 Knot per second) with the throttle closed and an increase in angle of attack until the stall occurs.

**Low wing** A low wing is one set low on the fuselage with the wing's lower surfaces flush, or almost flush with the bottom surface of the fuselage. Modern jet transports and business jets have their wings located under the fuselage – the main wing spar runs below, not through the fuselage – with belly fairings to smooth the airflow over them. In the 1930s, aerodynamic research and testing showed the low wing aircraft suffered from more drag and less lift compared to a high wing location. This is due to the wing's upper surface, low-pressure area interfering with the airflow along the sides of the fuselage. Fillets are designed to smooth out these two conflicting airflows and reduce the interference drag so produced.

*See **Fillets**.*

# M

**Mach, E.** Ernst Mach (1838–1916) was a physicist and philosopher of science. He was born in Turas, Austria and in 1864 he became the Professor of Mathematics at the University of Graz and a Professor of Physics in 1866 where his interest lay in optics and ballistics. In 1867, he became the Professor of Experimental Physics at Prague University where he stayed for the next twenty-eight years. During his time in Prague, he published over 100 technical papers. He died near Munich, Germany in 1916.

Pilots will remember Ernst Mach for his contribution to high-speed flight, in particular the supersonic region. Surprisingly, his interest in supersonic experiments started in 1873 when he studied shockwaves and secondary waves, etc. He was the first to show photographs of shockwaves using the schlieren photographic technique in 1887. Aerodynamicist, Jacob Ackeret (1898–1981) coined the now commonly used term Mach 1 (for the speed of sound) in 1929.

**Mach angle & cone** The Mach angle is the angle generated by the relatively weak shockwave. However, an aircraft is a three-dimensional body and in place of a Mach angle, there is a Mach cone or wedge. A Mach cone is formed at the aircraft's nose (like a boat's bow wave) where the angle is twice that of a Mach angle. Ernst Mach (1838–1916) demonstrated the existence of the conical shaped Mach cone with schlieren photography.

A Mach wedge is formed at the wing's leading edge. The line B-A and B-C (on *Diagram 57, Mach Angle & Cone*) at a tangent to the circles represents the Mach lines. The trigonometric sine angle of the cone angle 'B' is equal to the inverse of the Mach number where:

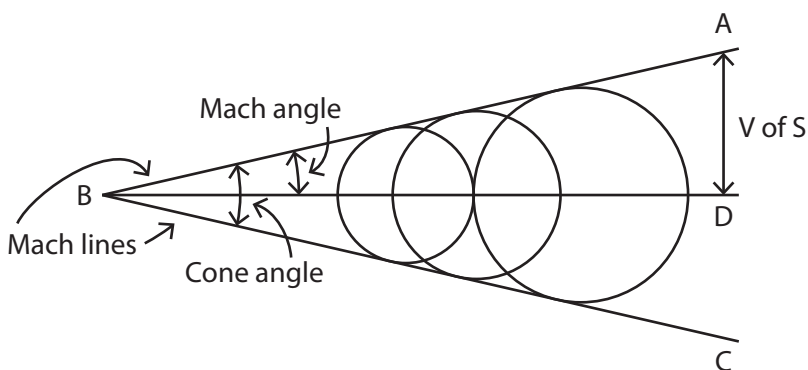
$$\sin 'B' = 1/M.$$

At Mach 1.0, a shockwave lies vertical to the surface and becomes an oblique shockwave as the Mach number increases higher than Mach 1.0, and the Mach lines converge reducing the Mach angle to an acute angle.

**Diagram 57, Mach Angle & Cone**, shows the speed of sound as vector A-D. The aircraft's velocity at a speed greater than Mach 1.0 is shown by vector B-D. Angle ABD is the Mach angle at a speed greater than Mach 1.0 in a two-dimensional flow. The Mach angle represents the angle between the direction of flight and the Mach line in a two-dimensional flow situation and is inversely proportional to the Mach number of the aircraft. The undisturbed free air stream outside of the Mach cone is known as the zone of silence.

The term Mach cone was coined not by Ernst Mach himself, but by Professor Ludwig Prandtl (1880–1953) and his co-workers in 1907.

See **Propagation of shockwaves**.



**Diagram 57, Mach Angle & Cone**

**Mach detachment speed** An aircraft flying at supersonic speed will initially form an attached shockwave on the aircraft's nose or wing leading edge. A further increase in Mach number to about Mach 1.2, will cause the shockwave to detach from the surface where it formed. On re-entry vehicles – the Space Shuttle and space capsules – the bow shockwave is designed to remain ahead of the surface, which is induced to form for cooling purposes. The speed at which the shockwave detaches and moves ahead of the surface is known as the Mach detachment speed, abbreviated as  $M_{DET}$ .

See **Detached shockwave; Blunt nose re-entry vehicle.**

**Mach diamonds** See **Shock diamonds.**

**Mach drag** See **Wave drag.**

**Mach intersection** The Mach intersection is the co-location of two shockwaves.

**Mach lines** Mach lines are not the same as shockwaves, they are weaker with smaller pressure changes. They form at the speed of sound, as opposed to shockwaves, which form at a slightly higher speed with greater amplitude.

Mach lines define the boundary limits on a Mach cone (see *Diagram 57, Mach Angle & Cone*) and also form in an expansion fan, see *Diagram 25, Expansion Flow & Fan*.

The supersonic flow cannot traverse a convex corner (an angle greater than  $180^\circ$ ) in one-step but must progress in stages through an infinite number of Mach lines known as the Prandtl-Meyer expansion fan, or wave. On passing through the fan, the flow velocity (Mach number) increases while the temperature, density and pressure all decrease, as expected in an expansion flow. The Mach lines in the fan are at ever-increasing angles to allow the flow to follow closely the surface curvature. [A subsonic flow would separate from the surface under similar conditions].

See **Expansion flow & fan.**

**Mach number (M or Ma)** Mach number is a dimensionless number defined as the ratio of the aircraft's true air speed to the speed of sound in the surrounding air. It is not the actual speed of the aircraft itself, it is a speed relative to the speed of sound. The Mach number varies as the square root of the absolute temperature. The formula is shown by the following:

$$M = \frac{V}{a}$$

Where M = Mach number

V = KTAS

a = Speed of sound at sea level =  $a_0 \sqrt{\theta}$

$\theta$  = Temperature ratio =  $T/T_0$ .

The Mach number may be expressed for example as Mach point 8, written as Mach 0.8, M = 0.8 or reversed as 0.8 M; all are acceptable terms.

Note the transonic range from Mach 0.8 to 1.2 overlaps the subsonic/supersonic junction of Mach 1.0. By convention, the following Mach numbers have been applied to speed regions from subsonic speeds to orbital velocity speeds:

|                  |                    |
|------------------|--------------------|
| subsonic         | less than Mach 1.0 |
| transonic        | Mach 0.8 to 1.2    |
| supersonic       | Mach 1.0 to 5.0    |
| hypersonic       | Mach 5 to 10.0     |
| high hypersonic  | Mach 10.0 to 25.0  |
| orbital velocity | Mach 25.0 +        |

The approximate speed of the aircraft in nautical miles per minute can be found by moving the decimal place in the Mach number, one digit to the right. Jacob Ackeret (1898 – 1981) coined the term Mach number in honor of Ernst Mach for his in-depth research into high-speed flows.



**Mach one (1.0)** Mach one (1.0) refers to the speed of sound. The speed of sound varies as the square root of the absolute temperature and decreases with altitude due to the decreasing temperature. At sea level, with an ISA temperature of +15°C, Mach 1.0 is equal to the following:

- 340.3 m/s
- 1225 KMH
- 761.2 MPH
- 661.7 Knots
- 1116.4 FPS.

An alternate name for the speed of Mach 1 is Mach unity.

*See Speed of sound (a).*

**Mach speeds** Mach speeds are reference speeds used by pilots, design engineers, and aerodynamicists. The following is a list of abbreviations with their meanings:

- $M_{CDR}$  = critical drag rise Mach number
- $M_{CRD}$  = critical drag Mach number
- $M_{CRIT}$  = critical Mach number
- $M_{DET}$  = detachment Mach number
- $M_{DF}$  = demonstrated diving flight Mach number
- $M_{DIV}$  = drag divergence Mach number. Same as  $M_{DR}$  or  $M_{CDR}$
- $M_{DR}$  = drag rise Mach number
- $M_{FS}$  = free stream Mach number
- $M_{FD}$  = force divergence Mach number
- $M_{IND}$  = indicated Mach number
- $M_{MO}$  = maximum operating Mach number
- $M_N$  = Mach number
- $M_{NE}$  = never exceed Mach number
- $M_{NO}$  = normal operating Mach number (archaic).

**Mach tuck** An aircraft not designed for supersonic flight can develop Mach tuck when it enters the transonic speed range and enters a steep, un-commanded dive.

As the air speed increases into the transonic range and eventually past the critical Mach number, the centre of pressure moves aft of the point of maximum wing thickness to about 50% MAC. The increased separation distance between the lift force and the centre of gravity intensifies the lift/weight couple causing the nose to pitch down or to 'tuck' (hence the term Mach tuck). The wing becomes aft loaded, which also adds to the pitch down. The downwash angle off the main wing decreases and flows over the tailplane reducing its angle of attack and hence, its downward force is reduced allowing the nose to pitch down; the effect is destabilising.

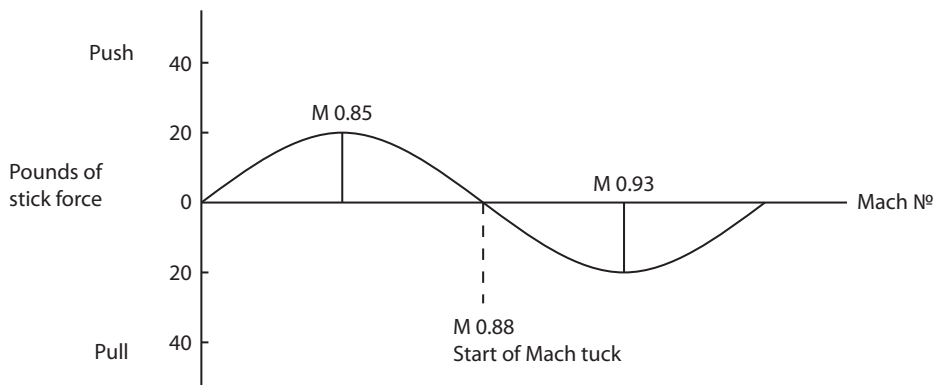
The associated shockwave on the main wing has the effect of changing the direction of airflow over the tailplane increasing its lift and raising the tail. This reduces the normal tail-down force allowing the plane to pitch nose-down and will require up-elevator pull to counteract it. It becomes a continuous cycle as the speed increases, the nose pitches down increasing the Mach number and nose-down movement. The aircraft is in a state of being speed unstable.

Another scenario occurs on swept wing aircraft when a shockwave located at the wing tip affects the centre of pressure moving it forwards causing the nose to pitch-up. Conversely, a wing root shockwave has the opposite effect, which moves the centre of pressure aft adding to the pitch down Mach tuck problem.

Mach meters have a red line speed limiting the maximum permissible Mach number known as the maximum operating Mach number ( $M_{MO}$ ). Modern jet transports have a built-in Mach trim system to automatically compensate for Mach tuck.

**Diagram 58, Mach Tuck & Stick Force**, shows the normal stick-push forward force with increasing speed up to the point where compressibility starts to take effect at about Mach 0.85. Stick-push force is then reduced to a point where it changes to a stick-pull

force during the Mach tuck stage of flight, around Mach 0.93. Many high-speed jet aircraft have Mach trimmers to counter the effect of Mach tuck.



**Diagram 58, Mach Tuck & Stick Force**

Mach tuck first made its presence known prior to WWII, as the new monoplane fighter aircraft types (Spitfire, Hurricane, etc) were able to reach speeds high enough to encounter the onset of Mach tuck. Recovery is very difficult due to the lack of either elevator authority or insufficient strength on the part of the pilot to pull the stick back and raise the nose against the forces involved. Many aircraft and pilots were lost due to Mach tuck during WWII and during test flying after the war, until the cause of it was understood.

Squadron Leader Tony Martindale, a test pilot for the Aerodynamics Flight at Farnborough, England, flew a Spitfire PR Mark XI on high-speed dives to research the effects of Mach tuck. During one dive from 40 000 feet, the aircraft entered Mach tuck with aileron reversal. It was not until he reached warmer, denser air at lower altitude that he was able to recover from the dive. During the dive, he reached Mach 0.92 (620 MPH) at 27000 feet, the fastest speed recorded in a piston-engine aircraft. In America, NACA was also busy studying the cause of Mach tuck around this time.

**Mach unity** Mach unity has the same meaning as **Mach one** (1.0).

**Mach wave/wavelet** Mach lines represent the angle at which Mach waves are formed, the faster the flow the smaller the angle. The tangent line on *Diagram 66, Propagation of Shockwaves*, represents the limit to which the Mach waves (pressure waves) have reached when the body (aircraft) has reached location 'D'. Mach waves are in effect, weak shockwaves.

*See Diagram 66, Propagation of Shockwaves.*

**Main flap** In a slotted flap system such as the Fowler flap, the main flap is the largest vane below the foreflap (and mid-flap if fitted).

**Mainplanes** An alternate name for the main wings of an aircraft. Mainplanes are mounted on the fuselage in the low-wing, mid-wing, or high-wing positions. An alternate location is to have the wing mounted on a pylon above the fuselage; it is then known as a parasol wing as found on the Consolidated PBY Catalina flying boat.

They are also known as **Airfoils**.

**Main rotor** The main lift and propulsive thrust-providing rotor disc(s) on helicopters have several different configurations of single or multi, main rotor discs as listed below:

- **Main, single rotor:** The Bolkow BK 117 has a single main rotor disc, mounted atop the mast directly above the engines, which is the most common layout for modern helicopters.



**Photo:** The Bolkow has a single disc, main rotor system.

- **Coaxial rotors:** Co-axial rotors are used where the two rotors are placed one above the other on the same drive shaft and rotating in opposite directions. The torque of each rotor cancels out negating the need for an anti-torque tail rotor and the dissymmetry of lift is cancelled out.



**Photo:** The Hiller XH-44 Hiller-copter was the first co-axial helicopter to fly in the USA, in 1944. This model is located in the Udvar-Hazy Centre in Washington, DC.

- **Intermeshing rotors:** These are twin rotors mounted closely, side-by-side with their rotor masts toed-out at an angle to each other in a transversely symmetrical manner allowing the blades to rotate in opposite directions and to intermesh. The rotor torque from each set of blades cancels out negating the need for an anti-torque tail rotor, and making the helicopter very stable. The German, Anton Flettner (1885 – 1961) developed the intermeshing rotor system on his Flettner Fl 282 anti-submarine helicopter. It is also known as synchropter.



**Photo:** The Kaman HH-43B Huskie has two intermeshing rotors mounted on individual pylons (removed in this photo). Pima Air & Space Museum in Tucson, Arizona.

- **Tandem rotors:** Tandem rotors are a set of counter-rotating rotors mounted at each end of the long fuselage on larger helicopters. The Boeing CH-47 Chinook and Sea Knight, for example use tandem rotors.



**Photo:** The Sikorsky HH-46 Sea Knight has a tandem rotor layout. This one is now homed in the USS Midway Museum, San Diego, Calif.



- **Transverse rotors:** Twin rotors mounted on out-rigger booms each side of the fuselage is known as transverse rotors.



**Photo:** The Bell Aircraft XV-3, experimental rotorcraft with a pair of transverse, tilt rotors mounted within the orange fairings shown here, is now resident in the National Museum of the US Air Force in Dayton, Ohio.

**Main wing** An alternate and more common name for the aircraft's mainplanes.

**Manoeuvre envelope** The manoeuvre envelope describes the parameters of air speed (V) limitations plotted in terms of CAS/RAS (calibrated/rectified air speed) versus the load factor (n) for each particular aircraft type. The envelope is therefore called a V-n (or V-g) plot or manoeuvre envelope.

The manoeuvre envelope below is based on:

- aircraft all up weight
- aircraft configuration (flap and gear position)
- Symmetry of loading – assumes aircraft is flying straight and level.

The manoeuvre envelope assumes the motion is only in the pitching plane. It does not take into consideration a combined pitch and roll manoeuvre where the 'g' force at each wingtip is different, being greater on one side and less on the other. Therefore, if the aircraft is rolling and pitching combined, the limit load factor is reduced by one third. Overstressing the aircraft will reduce its service life or even cause catastrophic failure of the airframe.

The load factor is located on the vertical axis of the V-n diagram and the air speed is located on the horizontal axis. Also included are the stall curve,  $V_S$ ,  $V_A$ ,  $V_C$ ,  $V_{NE}$ , and  $V_D$ . The positive limit load factor line is at 3.8g for normal category aircraft and the ultimate load factor at 5.7g. The negative limit load factor is at -1.52g and the negative ultimate load factor is at -2.28g.

The curved line from the zero 'g', zero speed position to the  $V_A$  manoeuvring speed limit at the junction of the 3.8g line represents the accelerated stall line or lift boundary, with flaps up at 1g load factor. The aircraft cannot operate in the speed range represented by the area on the graph to the left of the stall line – the aircraft's speed is too low and the aircraft stalls. The  $V_A$  manoeuvring speed line and maximum limit load factor junction is known as the 'corner speed'. This is where an aircraft achieves its maximum

manoeuvring performance. The true air speed is at the lowest value for maximum g-force without overstressing the airframe.

The manoeuvring speed ( $V_A$ ) is the only placarded speed, which reduces with weight because it is directly related to the stalling speed ( $V_{S1}$ ). At lighter weights, the stall line moves to the left, taking the manoeuvring speed  $V_A$  speed with it. Therefore,  $V_A$  for lighter weights is always less than the  $V_A$  found on the graph or given in the pilot's operating handbook. Hence the stalling speed decreases and therefore, the  $V_A$  decreases.

The junction of the maximum limit load factor and the red line  $V_{NE}$  limit on the envelope define the boundary of the buffet corner. The corner is angled off to ensure the maximum allowable load factor reduces with increasing indicated air speed. Flight is not permitted beyond that area, due to the possibility of encountering wind gusts, which could overstress the aircraft at that speed and load factor.

The maximum flap extension speed ( $V_{FE}$ ) is in line with the flaps-down limit load factor, which may be less than 3.8g. This corresponds to the top end of the white arc on the air speed indicator. The limit load factor with flaps down is reduced to less than the normal flaps up value; therefore,  $V_{FE}$  is usually lower than  $V_A$ .

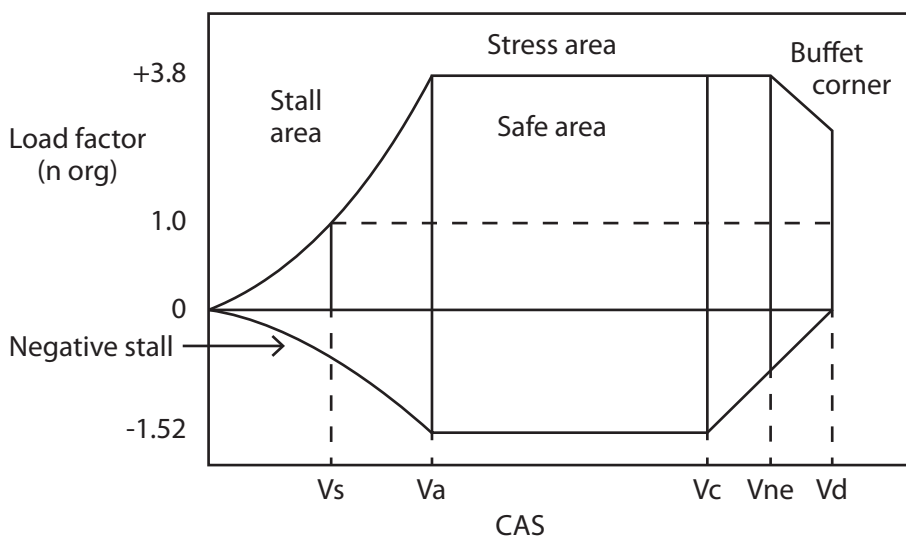
The maximum structural cruising speed ( $V_{NO}/V_C$ ) is located at the top of the green arc on the air speed indicator and is related to the load factors produced by the vertical and horizontal velocity gust gradients. The vertical gusts are the major problem because they alter the wing's angle of attack. An increase in angle of attack is akin to up-elevator being rapidly applied causing an overstress of the airframe. Turbulence should be avoided when flying above the  $V_{NO}/V_C$  speed, within the yellow arc on the air speed indicator. The aircraft may only be operated in the yellow arc in smooth air. It may be of interest to note here that turbine powered aircraft are certificated under different FAR rules to light piston-engine aircraft and do not include a yellow arc on the air speed indicator – the red line  $V_{NE}$  is at the top of the green arc. Pilots of light aircraft should seriously consider the risk of operating above the maximum structural cruising speed ( $V_{NO}/V_C$ ) at the top end of the green arc at

any time. If turboprop aircraft avoid this region, light aircraft pilots should take heed! Exceeding the positive or negative ultimate load factor can result in structural airframe damage. The aircraft could be overstressed in flight!

The never exceed speed limit air speed or  $V_{NE}$  (red line) is a design point based on the flaps up stall speed multiplied by the square root of the limit load factor. It is located at the top end of the air speed indicator's yellow arc and this is the aircraft's maximum safe, permissible speed limit at any time. Flight above this speed may incur structural damage or failure due to encountering a critical wind gust or stress due to aileron reversal, divergence, flutter or compressibility problems (Mach tuck) etc. At a speed above the  $V_{NE}$  (red line) there is a safety margin terminating in the  $V_D$  design dive speed – the ultimate limit air speed. The safety margin is the domain for test pilots only; structural failure is guaranteed above  $V_D$ !

The manoeuvre envelope is also known as the flight envelope.

See **Gust envelope; Limit & ultimate load factors.**



**Diagram 59, Manoeuvre Envelope**

**Manoeuvre margins** The manoeuvre margins are measured from two positions as a percentage MAC from the stick-fixed and stick-free positions to the centre of gravity position.

Sidney B. Gates (1893–1973) introduced the manoeuvre margin to aerodynamics while working at the Royal Aircraft Establishment at Farnborough, UK.

*See Diagram 63, Pounds of Pull V. CG Percentage MAC.*

**Manoeuvre point** The centre of gravity location is normally considered the point where the aircraft rotates about all three axes, but to be precise the aircraft rotates about the manoeuvre point. The manoeuvre point has two positions depending on whether the aircraft is flying stick-free or stick-fixed and is not necessarily co-located with the aircraft's centre of gravity. The zero stick force per 'g' with either stick-free or fixed, determines the manoeuvre point's location.

Sidney B. Gates (1893–1973) introduced the term manoeuvre point.

*See Aerodynamic axes.*

**Manoeuvre speed ( $V_A$ )** The manoeuvre speed  $V_A$  has been commonly stated as the 'maximum speed at which an abrupt and full control deflection can be made without overstressing the aircraft'.

The  $V_A$  manoeuvring speed is related to the limit load factor and  $V_{S1}$ , the flaps up stall speed where  $V_A$  for normal category aircraft equals the square root of 3.8 times  $V_{S1}$  or nearly twice the stall speed. The manoeuvring speed is stated in the aircraft's flight manual for the maximum all up weight and limit load factor. At aircraft weights below the maximum all up weight the  $V_A$  speed decreases. Flying at a speed below  $V_A$  and inducing a rapid pitch up will stall the aircraft and relieve the loading before the aircraft is overstressed. A rapid pitch up when flying above the manoeuvring speed could lead to overstressing of the airframe.

The manoeuvring speed is the maximum speed to be used when flying in turbulence to avoid structural damage caused by wind-gust induced loads.

The manoeuvre speed can be calculated from the following:

$$V_A = V_{S1} \sqrt{n \text{ limiting 'g'}}$$

Where  $V_A$  = manoeuvre speed

$V_{S1}$  = stall speed flaps up

$n$  = limit load factor.

It is also known as the design manoeuvre speed.

See **Manoeuvre envelope**.

**Manoeuvring stick force gradient**      See **Stick force gradient**.

**Maskell, E.** Eric C. Maskell was a British aerodynamicist based at the Royal Aircraft Establishment (Farnborough), England. In collaboration with Dr. Dietrich Küchemann (1911–1976), he made a large contribution in researching the effects of flow separation over aircraft wings, amongst many other achievements.

**Mass** Mass is a scalar quantity and defined as:

- the quantity of matter in a body – volume times the density of the object
- mass is a measure of inertia – or how difficult it is to start or stop a body in motion or change its direction of motion
- mass is directly proportional to inertia – greater mass has greater inertia
- the mass of a body remains constant, as opposed to its weight, which decreases with increasing distance from the centre of the earth
- mass is the sum total of all the atoms (number of protons and neutrons) in the body

- The SI unit for mass is the Kilogram (Kg) or the now obsolete British 'slug'.
- One slug = 14.59 kg. [Note, weight is measured in pounds or Newtons].

One Newton = 1 kg m/sec<sup>2</sup>

1 pound = 1 slug ft/sec<sup>2</sup>.

Although mass is normally defined in terms of inertia, it is measured by gravitation. Newton's Laws of Gravitation published in his book *Principia* in 1687 states 'the force of gravity between any two bodies attracts each other with a force proportional to the product of their masses and inversely proportional to the square of the distance between them'. Mass is shown as:

Mass = weight/g, or conversely, weight = mg.

The mass, density and volume of a fluid are related thus:

$$m = \rho v$$

Where m = mass

$\rho$  = density

v = volume.

Compare with weight, which is a measure of the Earth's gravitational pull, and thus varies with distance from the centre of the Earth, whereas mass is constant.

**Mass balance** A mass balance is used to prevent torsional flutter (or vibration) of a control surface. The mass, usually located internally, is located ahead of the hinge line axis to place the centre of gravity at, or forward of the hinge line. An aircraft with inset hinges or horn balance may have the mass located within the control surface or alternatively, it may be located on an arm extending forward of the hinge line as in the photograph.

**See Aileron reversal; Flexural aileron flutter; Torsional aileron flutter; Forward sweep wings; Flutter; Aeroelasticity; Divergence; Divergence speed.**



**Photo:** An external mass balance mounted on the ailerons of a DeHavilland 114A Heron aircraft.

**Maximum camber** The maximum camber is the point on the mean camber line where it is at its maximum displacement above the chord line. The location of the maximum camber determines the airfoil's characteristics; for example, at 25% MAC the wing can produce a very high lift force with good low-speed stability. With the maximum camber further rearwards at around 40–50% MAC, the wing experiences greater laminar flow and improved high-speed characteristics.

See **Camber and camber line; Mean camber line.**



**Maximum lift coefficient** The maximum lift coefficient ( $C_{Lmax}$ ) is achieved at the point just before the wing stalls. When plotted on a graph as a well-rounded curve, it indicates docile stall behavior. A sharply peaked curve indicates a more abrupt stall break.

See *Diagram 49, Lift Coefficient V. Angle of Attack; Lift coefficient; Critical angle of attack.*

**Maximum thickness** The maximum thickness is the overall dimension of the thickest part of an airfoil. Greater thickness produces greater lift for comparatively low-speed airfoils; high-speed airfoils favor wings with a small maximum thickness.

**Maximum thrust differential** The maximum thrust differential represents the difference between the maximum thrust pounds available from the propeller and thrust pounds required from the propeller. It corresponds to  $V_x$  the maximum angle of climb speed.

See *Diagram 60, Thrust & THP Required & Available Curves.*

**$M_{CDR}$**  This is the abbreviation for critical drag Mach number. It is the speed – or Mach number – at which, the drag coefficient starts to increase due to compressibility.

See **Drag divergence curve.**

**$M_{CRIT}$**  The abbreviation for the critical Mach number of an aircraft's wing.

See **Critical Mach number.**

**$M_{DIV}$**  When the aircraft's velocity reaches the point where the drag coefficient has risen by 0.002 while maintaining a constant angle of attack, the speed of  $M_{DIV}$  (the drag divergence Mach number) has been reached.

**Mean aerodynamic chord (MAC)** The mean aerodynamic chord (MAC) is drawn through the geographical centre (centroid) of the wing's plan area. The mean aerodynamic chord is the mean width of the wing (not the average width), which is usually referred to as percentage MAC.

It is a common chord-weighted measurement used for calculating aircraft pitching moments and longitudinal stability calculations. Furthermore, the mean aerodynamic chord of the nett wing area is used to find the Reynolds number when calculating wing drag.

The MAC is related to the aircraft's longitudinal stability and centre of gravity calculations. MAC therefore, does not have the same meaning as average chord.

*See Aerodynamic chord; Chord.*

**Mean blade width ratio** The ratio of the propeller's blade width to the propeller's diameter is known as the mean blade/width ratio.

**Mean chord** The mean chord is calculated by dividing the gross wing area by the wingspan, or the average of the wing root and tip chords.

*See Chord; Mean aerodynamic chord (MAC); LEMAC.*

**Mean camber line** The mean camber line is located midway between the wing section's upper and lower surface from the leading edge to the trailing edge, indicating the curvature of the airfoil section. A highly cambered airfoil section has greater curvature than a section with a relatively flat upper surface. No camber exists on a symmetrical airfoil. The camber line is also known as the mean line, or median line.

*See Diagram 4, Airfoil Terminology; Camber and camber line; Maximum camber.*

**Mean geometric pitch** The mean geometric pitches of all the propeller's blade elements from the root to the tip of the blade.

*See Geometric Pitch.*

**Mean line, or median line** *See Mean camber line.*

**Measurement of power** The power of a piston-engine can be measured in various ways such as indicated horsepower, brake horsepower, thrust horsepower, etc. The Metric system of Kilowatts (kW) is used in some countries.

Indicated horsepower is calculated from the area of an indicator diagram and RPM being dependent on the dimensions and number of cylinders. As pilots, we are not interested in indicated horsepower; that is left to the engine designers. Of more concern to pilots is the brake horsepower (BHP) that governs aircraft performance. Brake horsepower is the power produced by the engine at the output shaft or, in the case of an aircraft, the power available to the propeller. Brake horsepower is always less than indicated horsepower due to friction losses and power required to drive the engine's ancillary equipment. The power produced by the propeller, the thrust horsepower, is around 70–90% of the brake horsepower due to propeller deficiencies. The word 'brake' as in brake horsepower is taken from the dynamometer's alternate name of 'prony brake'. The mechanical efficiency of the engine is equal to the ratio of brake horsepower to indicated horsepower.

Reference to engine brake horsepower can be expressed in Imperial or SI units as follows:

$$1 \text{ HP} = 550 \text{ foot-lb./second} = 33000 \text{ foot-lb./minute} = 0.7457 \text{ kW}$$

The Imperial unit for power is the Watt symbol 'W' or the Kilowatt

$$1 \text{ Watt} = 1 \text{ Joule per second.}$$

The formula for power is:

$$\text{Power expended (W/t)} = \frac{W \text{ ft. lb}}{550 \times t \text{ (seconds)}}$$

Where W = work done in Joules

t = time required to do work in seconds.

**Mechanics** The oldest branch of physics is the study of mechanics. It involves applied mathematics in the study of statics, dynamics, friction, turning forces, moments, and fluid mechanics, etc.

**Mechanical Energy** Mechanical energy is the sum of the potential and kinetic energy. The amount of energy a body can impart is measured by the amount of work the body can do. In other words, the quantity of work done by a body depends on the energy due to its motion. It can be seen work and energy are closely related. The unit for measuring energy is the same as for work, that is, the Joule. This transfer of energy is known as the 'conservation of energy'.

See **Potential & kinetic energy; Conservation of Energy.**

**Mid-flap** The middle airfoil vane in a triple-slotted flap system.

**Mid-wing** A mid-wing as the name implies – is set half way up the fuselage as found, for example, on the Piper Aerostar and the WW II North American B25 Mitchell. This is a more aerodynamically efficient location than either a low or high wing position.

See **Belly fairing.**



**Photo:** The North American B-25 Mitchell J, displays the mid-wing design of this type. This example is on display in the Pima Air & Space Museum in Tucson, AR.

**Minimum control speed** The minimum control speed ( $V_{MC}$ ) is the speed at which a multi-engine aircraft can fly with a failed engine and still maintain directional control.

The force from the rudder to keep the aircraft flying straight counteracts the yaw force caused by an engine failure. Below a certain speed, rudder authority is reduced and the aircraft will diverge from the required heading. The speed at which this occurs is known as the 'minimum control speed' or  $V_{MC}$ . On a conventional twin-engine aircraft, the yaw force is greatest with the critical engine failed. In effect, there are two different air speeds at which the rudder fails to maintain directional control. In other words, there is a  $V_{MC}$  for each engine. In practice, the higher of the two air speeds is taken as the operational  $V_{MC}$  air speed. Aircraft with counter-rotating propellers have the same amount of yaw

force with either engine failed; therefore, the  $V_{MC}$  is the same for either engine.

The minimum control speed can be further defined as  $V_{MCA}$  for use when the aircraft is airborne. The  $V_{MCA}$  should be no higher than 1.2 times the stalling speed. On the ground, the minimum control speed is defined as  $V_{MCG}$ , which must be lower than the take-off decision speed ( $V_1$ ).

**Minimum drag speed** To most pilots it is common knowledge that the thrust equals the drag. What is not so obvious at first is the fact that minimum drag is found at a speed slightly higher than the minimum power speed. Minimum drag is found where the decreasing induced drag is equal to the increasing parasite drag as shown by **Diagram 20, Total Drag Curves**, at 1.3 times  $V_{S1}$ . In addition, **Diagram 60, Thrust & THP Required & Available Curves**, shows the minimum drag to be at the  $V_Y$  best rate of climb speed, slightly higher than  $V_X$  best angle of climb speed.

In the minimum drag formula below, the true air speed cancels out leaving drag. This is the minimum drag speed, also known as the drag/velocity ratio (D/V ratio). The reciprocal of this ratio is the maximum velocity/drag ratio (max V/D ratio) which is what we are interested in here. Another name for this is the more familiar maximum lift/drag ratio (max L/D ratio) which occurs at a power setting of approximately 40% BHP. The speed also coincides with the maximum rate of climb speed and the best glide ratio speed, around 70–80 Knots for a typical light aircraft. At the lower power setting of 40% BHP, the first definition of range speed will be satisfied that is, ‘the least amount of fuel used per unit distance’.

$$\frac{\text{Power}}{\text{TAS}} = \frac{\text{Drag} \times \text{TAS}}{\text{TAS}} = \text{minimum drag}$$

See **Lift/drag ratio; Diagram 20, Total Drag Curves; Diagram 60, Thrust & THP Required & Available Curves.**

**Minimum radius of turn** An aircraft's maximum turning performance and its ability to make a turn with a minimum radius is defined by the three factors of aerodynamics, structural limit, and the thrust or power limits, thus:

- The aerodynamic limit: produced by the wing loading and maximum lift coefficient.
- The structural limit is dependant on the maximum limit of the manoeuvring load factor to avoid structural damage.
- The thrust or power limits determine the maximum thrust required for maintaining a constant rate of turn at constant altitude with the wings developing the maximum load factor and induced drag.

*Diagram 59, Manoeuvre Envelope*, shows how the aerodynamic and structural limits define the maximum turning performance available. When the aircraft is flying at the stall speed, the lift coefficient is at a maximum and there is only sufficient lift to hold the aircraft in the air and no excess lift available for turning. This is shown by the aerodynamic limit line being in contact with the stall line on the left side of the diagram. An increase in velocity above the stall speed increases the maximum lift to become greater than the aircraft's weight. The load factor and bank angle increase until eventually the speed increase limits the minimum radius of turn at the manoeuvre speed.

The manoeuvre speed marks the intersection between the aerodynamic limit and the structural limit becomes the limiting factor – to avoid overstressing the aircraft – for minimum rate of turn with angle of bank remaining constant. This is also known as the corner speed, where the maximum lift is available for a minimum radius turn. Above this speed, the turn radius varies with the square of the KTAS.

It is also known as the tactical turning performance.

See **Manoeuvre speed ( $V_A$ ); Manoeuvre envelope**.

**Minimum static margin** The static margin is the variable distance from the aircraft's actual centre of gravity location to the neutral point. However, the minimum static margin is the fixed distance from the aft centre of gravity location to the neutral point. The minimum static margin as an alternate name for the safety margin.

See *Diagram 12, Centre of Gravity Range*.

**Mission adaptive wing** A mission adaptive wing maintains peak aerodynamic efficiency under almost all flight conditions. The wing is made of composite flexible materials, which allows it to change the shape of its camber, leading, and trailing edges in flight to suit the requirements of the manoeuvre being performed. On-board computers are used to vary the surface contours of the



**Photo:** A General Dynamics F-111 similar to this example was converted to test the mission adaptive wing principle. This aircraft is a resident of the RAF Museum at Cosford, England.



wing – lowering and raising the ailerons, flaps, or leading edge devices as required to alter the contour of the wing to effect air manoeuvres.

The mission adaptive wing has the advantage of reducing aerodynamic drag and aircraft structural weight, allowing a higher aspect ratio wing to be used and a greater choice in wing thickness, wing sweep angle, and wingspan. These factors reduce the aeroelastic twisting of the wing from conventional aileron input, avoiding control reversal. In fact, it is the wing itself that creates the required control forces to manoeuvre the aircraft as it reacts to the energy in the airstream over the wings.

The mission adaptive wing was flight-trialed using a General Dynamics F-111 research aircraft. The NASA Dryden Flight Research Centre also studied the mission adaptive wing concept when they used a McDonnell Douglas F/A-18 Hornet, which was converted by Boeing to test what they called the Active Aeroelastic Wing concept. The results of the research were positive proving jet fighter aircraft of the future would greatly benefit from this new type of wing for greatly enhanced manoeuvrability.

It is also known as aeroelastic tailoring.

**Moment** The moment is the product of the weight or force multiplied by the arm (the perpendicular distance from the datum) expressed in inch-pound, or foot-pound, or Newton-meter (N-m). Increasing the arm will increase the power of the moment. Clockwise acting moments are considered positive as opposed to anti-clockwise moments, which are termed negative.

The term moment has common usage in aerodynamics where it describes an aircraft's tendency to rotate around the OY lateral axis; it is positive when the aircraft rotates nose-up and negative when it rotates nose-down. The moments of an airfoil vary with changes in angle of attack except when the centre of pressure and aerodynamic centre are at the 25% MAC location for a symmetrical airfoil, in the unstalled condition. For a cambered airfoil, it is the theoretical location and not necessarily the actual location.

The units for moments are the same units as used for couples.

See **Principal of moments; Torque force.**

**Moment arm** The moment arm is the horizontal distance in inches or millimeters measured from the reference datum to a given location. For all moments aft of the datum, the algebraic sign is plus (+) and all moments forward of the datum are negative (-). For example, a moment arm of 40 inches aft of the datum is equal to 'station 40' (the plus sign can be omitted for positive signs).

*See Arm; Fuselage station.*

**Moment coefficient** The moment coefficient ( $C_M$ ) is a non-dimensional number divided by the dynamic pressure. The moment coefficient may be in the form of the pitching moment coefficient  $C_M$ , the aerodynamic centre moment coefficient  $C_{MAC}$ , or the centre of gravity moment coefficient  $C_{MCG}$ . The reference moment coefficient occurs at zero degrees angle of attack.

Reynolds number has no affect on the moment coefficient except at high angles of attack.

**Momentum** Momentum is the product of mass times velocity ( $M = mv$ ). Momentum is the quantity, or measure of motion of a moving body, or the driving force (impetus) gained due to movement. Because velocity is in the equation, momentum has direction and because velocity is a vector, then momentum is a vector quantity. The direction of momentum acts in the direction of velocity. It follows; a stationary aircraft (body) does not have any momentum although it does have inertia.

Momentum is defined as  $M = mv$

where:  $M$  = momentum of the object

$m$  = mass of the object (kg or lb)

$v$  = velocity of the object (m/sec or ft/sec).

Inertia is related to Newton's 1<sup>st</sup> Law of Motion and applies to a stationary body, or a moving body in a straight line at a constant speed. Newton's 2<sup>nd</sup> Law of Motion is related to momentum rather

than acceleration and is stated, as ‘the rate of change of momentum is equal to the net force being applied’.

We can also relate momentum of a moving body to Newton’s 3<sup>rd</sup> Law of Motion. ‘When a body receives momentum in a collision in a given direction, the other body will receive an equal amount of momentum in the opposite direction.’ Consider the case of two billiard balls colliding, where they both rebound away from each other.

Greater momentum in a body (aircraft) means it is harder to stop its motion. When it has greater mass or velocity, it will also have greater momentum. A large aircraft with its greater mass has greater momentum and therefore requires a greater landing distance in order to come to a stop. Momentum is not to be confused with kinetic energy or inertia. With an increase in air speed, momentum increases while the inertia remains constant.

Note: carefully compare the two definitions of Newton’s 2<sup>nd</sup> and 3<sup>rd</sup> Laws of Motion. Compare also, the symbols for momentum =  $mv$  and energy =  $\frac{1}{2}mv^2$ . The SI unit for momentum is kg m/s. This is the same as for mass and velocity.

*See Axial momentum theory; Energy – potential & kinetic.*

**Momentum drag** Momentum drag is caused by the difference between the momentum of the air entering a jet engine’s intake and the aircraft’s forward speed.

When the airflow enters the divergent passage in front of the jet engine’s compressor fan the flow-speed decreases abruptly and the static pressure rises. It is the change in the airflow’s momentum as it enters the intake that causes the momentum drag.

It is also known as intake momentum drag or ram drag.

**Momentum exchange** When the turbulent boundary layer over the wing mixes with the high-speed outer layers of the laminar flow, the turbulent flow-speed must increase to match that of the laminar layer above it. The increase in speed, known as the momentum exchange, results in an increase in boundary layer drag.

**Momentum theory** The momentum theory deals with the aerodynamic workings of propellers as introduced by Rankine in 1865 and Froude in 1885.

It is also known as the disc actuator theory.

*See* **Axial momentum theory.**

**Momentum vortex blade element theory** *See* **Blade element theory.**

**Monoplane** A monoplane is an aircraft (airplane or glider) with only one set of main wings as found on nearly all modern aircraft. The term monoplane was introduced in 1907.

The first monoplane was built by a Romanian inventor Trajan Vuig in which he flew a short distance of twelve metres (40 feet) on 18 March 1906. With more success, the Frenchman Louis Bleriot (1872–1936) became the most successful designer and builder of monoplanes in the early 1900s. His most notable flight was the first to cross the English Channel in his Bleriot Type XI in 1909.

**Motion** There are two kinds of motion applicable to aerodynamics:

- linear motion is motion in a straight line
- curvilinear motion is motion on a curved path.

**Moustaches** An alternate name for canard foreplanes is moustaches, mounted on the forward fuselage.

*See* **Canard wings.**

**Movables** An alternate name for primary and secondary flight controls is the term movables.

**Mu ( $\mu$ ) ratio & barrier** The Mu ratio is the ratio of a helicopter rotor blade's tip speed to the craft's forward speed ( $V_{\text{TIP}}/V_A$ ). Most helicopters operate with a Mu ratio of around 0.3, where the rotor tip speed is three times the craft's forward speed. It is also known as the rotor tip advance ratio.

The Mu barrier is a ratio of Mu one (Mu 1) where the rotorcraft's rotor tip speed is equal to the craft's forward speed. In this limiting condition, the retreating blade experiences total airflow reversal over the whole length of the blade, which causes instability in the rolling plane and limits the forward speed of all helicopters.

The Greek term Mu ( $\mu$ ), pronounced as Mew or Moo, is a term common to rotorcraft engineering.

**Munk, Dr. M.** Dr. Maxwell Munk (1890–1986) was born in Hamburg, Germany. He was educated at the Hanover Technische Hochschule before studying aerodynamics at the University of Gottingen. He emigrated to the USA in 1920 where he then joined NACA Langley in Hampton, Virginia.

During his time with NACA, he explained the reason for the difference in wind tunnel data and full-scale test data. He claimed the difference in the air density in the wind tunnel is different to the actual density experienced by an aircraft in flight. He solved the problem by introducing the Reynolds number to adjust the data to its true value and suggested that airfoils should be tested in variable density wind tunnel, which he invented, designed, and supervised its operation at the Langley Research Centre in 1923 where the resulting data collected was far more accurate. Dr. Munk also introduced the thin airfoil theory, the linear theory of aerodynamics, further research on biplanes initiated by Professor Albert Betz (1885–1968), and many more contributions to aerodynamics.

**Mushing** Mushing refers to the un-commanded rate of descent of an aircraft. When flying at high angles of attack/low air speed, on the backside of the power curve (or region of reverse command) a high sink rate may develop – the aircraft mushes.

Mushing can also occur when up-elevator is applied to produce a pitch-up and the aircraft does not respond as required but continues along the same flight path – it mushes along.

# N

**NACA/NASA** The National Advisory Committee for Aeronautics (NACA) was established on 3 March 1915. The purpose of NACA and now the National Aeronautics and Space Administration (NASA) is to carry out extensive aeronautical research in aerodynamics, propulsion systems, and rockets and to oversee the space program. For nearly one hundred years, NACA/NASA has maintained its world leadership in civil aeronautics and space exploration by working with USA universities, aerospace companies and other Federal agencies.

NACA received a name change to NASA on 10 October 1958. The Head Quarters for NASA is located in Washington D.C. USA, which administers several research centres scattered around the USA.

Dr. George Lewis (1882–1948) was one time director of NASA followed by Dr. Hugh Dryden (1895–1965). Dr. Joseph S. Ames (?–1943) was the chairperson circa 1928. All three men now have research centres named in their honor.

**NACA cowling** Fred Weick (1899–1993) in 1928, a NACA engineer at Langley Research Centre, introduced the NACA engine cowling.

Fred Weick discovered the fact that an air-cooled radial engine surrounded by a streamlined wide-chord shroud reduced the cooling drag by about two thirds and improved engine cooling airflow pressure. As with any drag reduction, it improves aircraft performance by increasing the speed and lowering fuel consumption allowing extended range.

A Curtiss AT-5A trainer aircraft was used to flight test the new NACA cowling in 1928, with a recorded increase in cruise speed from 118 MPH to 137 MPH. The Lockheed Vega was one of the first aircraft to benefit from the new cowling. NACA was rewarded

with the first of its five Collier Trophies for its engine cowling design in 1929.

Compare the NACA cowling with the British designed Townend Ring, a similar type of engine cowling.

It is also known as a pressure cowling.



**Photo:** The Douglas O-38F equipped with a NACA cowling for reduced drag and improved engine cooling. This aircraft was photographed at the National Museum of the US Air Force in Dayton, Ohio.

**Nacelle** A nacelle is a streamlined compartment, commonly surrounding an engine (or other protruding aircraft part).

**National Physical Laboratory** The National Physical Laboratory is located at Teddington near London, England. It was established in 1900 and is still in operation today.

Thomas Stanton established a small wind tunnel there as early as 1903. In 1912, the first of several larger wind tunnels were built.



A more advanced Duplex tunnel was built in 1918; this comprised of two, seven by eight feet wind tunnels combined into one unit seven by fourteen feet long.

During the design and development of Concorde, around one hundred different wing designs were tested at this laboratory, as also was the bouncing bomb (in the water tanks) of Dambuster fame.

**Natural laminar flow airfoils** An alternate name for laminar flow airfoils. High-tech computers are used to design natural laminar flow airfoils for greater performance at higher cruise speeds.

See **Laminar flow airfoils**.

**Navier-Stokes Equations** Frenchman Claude-Louis Navier (1785–1836) and Englishman George Gabriel Stokes (1819–1903) independently developed the Navier–Stokes scalar equation (now used in aerodynamic calculations) in the first half of the 19<sup>th</sup> Century, circa 1840. The equation is derived from the continuity, energy, and momentum equations associated with the conservation of momentum, mass, and energy.

The Navier – Stokes Equation is a set of coupled differential equations that describe the influence of temperature, pressure, density and velocity on an unsteady, three-dimensional incompressible, viscous flow (Newtonian fluid) as found in a boundary layer.

The equation, which is far too complex for this book, is an extension of the earlier Euler Equations. The boundary layer exists because of the fluid viscosity. The aerodynamicist must predict the characteristics and drag produced by the boundary layer and hence, its effect on the design and performance of the aircraft in question. Today, computational fluid dynamics solves the complex mathematical problems with the use of high-speed computers.

**Negative camber** *See Reflex camber airfoil.*

**Negative g** An aircraft pitching nose-down causes a negative 'g' force. Negative 'g' acts in the opposite sense to a normal positive 'g' pitch-up. The pitching movement is in the vertical plane where the aircraft pitches around the OY lateral axis.

A nose-down pitch movement into a dive, inverted flight, or flight in turbulence can all induce a negative 'g' force on the aircraft. Negative 'g' is uncomfortable for most pilots to tolerate and the manoeuvre is not as common as normal positive 'g'; therefore, normal category aircraft are stressed to a negative limit load factor of minus 1.52g, which is less than the positive limit load factor of +3.8g.

*See Manoeuvre envelope; Limit & ultimate load factors.*

**Negative pressure gradient** A decreasing or favourable pressure gradient is known as a negative pressure gradient.

*See Favourable pressure gradient.*

**Negative propeller thrust** Propeller thrust acting in the reverse direction to provide braking action after landing.

*See Reverse thrust forces.*

**Negative stability** Negative stability in an aircraft is undesirable; it refers to instability.

**Negative stagger** An aircraft with negative stagger has the lower wing placed forward of the upper wing. Negative stagger is not as common as forward stagger. The Beech Model 17 Staggerwing is a good example of an aircraft with negative stagger.

See **Stagger; Biplane interference; Orthogonal biplane.**



**Photo:** The Beech 17 Staggerwing is one of the most popular biplanes ever built and has the unusual wing arrangement of negative stagger. The top wing is located further aft than the lower wing – the opposite way to most biplanes.

**Negative stall** A negative stall is a relatively rare manoeuvre compared to a normal (positive) stall on an aircraft wing, but it can happen. The V-n manoeuvre envelope shows the negative stall line in the lower, left corner of the diagram.

It can be of concern when acting on the pressure surface (the under surface) of axial turbine blades within a gas turbine (jet) engine. It is also known as a negative incidence stall.

See **Diagram 59, Manoeuvre Envelope.**

**Negative sweep** Negative sweep is an alternate name for a forward sweep wing.

*See Forward sweep wings.*

**Negative thrust** *See Reverse thrust forces.*

**Negative torque system** The system senses negative torque when the engine is generating power and adjusts the propeller blade angles to a more coarse setting.

Under certain flight conditions, with the engine throttled back to idle and the propeller in fine pitch, the propeller may produce zero thrust, or drag. The drag in this condition is known as 'flat plate drag'. The propeller disc can be considered as being a large flat solid plate. Turboprops in particular, with their large diameter props are susceptible to drag under this condition, with the fixed-shaft turboprop being affected more than the free-shaft turboprop. The constant-speed unit would naturally select a fine pitch setting, but this produces the highest drag and negative torque with the engine idling. To alleviate the high drag and negative torque at low power, turboprop engines employ a 'negative torque system' incorporated within the constant-speed unit/reduction gear assembly. The negative torque system senses when negative torque is being produced by the propeller and commands the constant-speed unit to turn the blades to a more coarse pitch setting. The blade angle will be at the correct setting to absorb a predetermined amount of horsepower. As soon as the negative torque is removed with increased power selection, the constant-speed unit resumes normal operation.

**Negative yaw** A negative yaw is considered to exist when the aircraft rotates around its OZ normal axis to the left.

*See Axes; Aerodynamic axes; Normal axis.*

**Net thrust** Net thrust is equal to gross thrust minus the aerodynamic drag of the fuselage, or engine nacelle behind the propeller. It is an in-flight measure of thrust as opposed to static thrust, which is measured on the ground.

*See Thrust terminology.*

**Neutral point** The neutral point of the aircraft is determined by the area of the stabilizer and its distance (lever arm) from the main wing. When the aircraft's centre of gravity is moved rearwards, static stability is reduced until a point is reached where it becomes neutral. This point is known appropriately as the neutral static stability point (or the neutral point for short) and has several significant features related to the centre of gravity location.

When the centre of gravity and the neutral point are co-located, neutral stability exists, and the pitching moments will remain the same at all angles of attack. Increased wing lift moments will equal the tail lift moments resulting in zero pitching moments to restore stability.

The neutral point is also the location from which the static margin is measured. This is the variable distance from the neutral point to the actual centre of gravity location expressed as % MAC. At a forward centre of gravity the static margin and hence, the stability is the greatest. However, as the centre of gravity moves aft, it decreases the static margin and the static stability. Therefore, stability is proportional to the static margin.

If the centre of gravity is located forward of the neutral point, the aircraft's static stability will be positive, but the static margin will be negative. It follows, when the centre of gravity is aft of the neutral point, the aircraft will suffer from longitudinal instability, although the static margin is positive. The aircraft will be very difficult and dangerous to fly in this condition. For a tailless aircraft to have longitudinal static stability the centre of gravity must be located forward of the aerodynamic centre, itself co-located with the neutral point.

Variations in power settings, flap and undercarriage positions during flight, all produce changes in the aircraft's pitching moments. Therefore, the difference in pitching moments between a full power climb and descending at idle power will cause the neutral point to vary its position by 7–8% MAC. Due to this variation in position, albeit by only a small amount, the centre of gravity aft limit is always forward of, and separated from the neutral point by the safety margin.

The safety margin is the *fixed* distance from the neutral point to the aft centre of gravity position and the static margin is the *variable* distance from the neutral point to the aircraft's centre of gravity location.

Sidney B. Gates (1893–1973) while working at the Royal Aircraft Establishment at Farnborough, UK introduced the neutral point to aerodynamics.

*See Diagram 12, Centre of Gravity Range; Diagram 78, Static Stability V. Angle of Attack.*

**Neutral stability** A correctly loaded aircraft with the centre of gravity located within its limits will have positive stability. Moving the centre of gravity aft to the neutral point, at approximately the 40% MAC position, the stability will change from positive to neutral. Any disturbance in the pitching plane will incur greater oscillations and take longer before the upset is damped out.

*See Stability; Diagram 78, Static Stability V. Angle of Attack.*

**Never exceed speed ( $V_{NE}$ )** The never exceed speed ( $V_{NE}$ ) as the name implies, should never be exceeded by an aircraft in flight under any circumstances. Exceeding the never exceed speed may overstress the aircraft leading to possible airframe structural failure. The never exceed speed is marked on the air speed indicator by a red line at the top end of the yellow arc. Flight up to the never exceed speed should be carried out with caution in smooth air only, due to possible airframe overstress if wind gusts

are encountered. The  $V_{NE}$  speed is set by the aircraft manufacturer at 90% of the design dive speed ( $V_D$ ).

See **Manoeuvre envelope**; **Gust envelope**.

**Newton (force)** The Newton (N-m) is a unit of force acting for one second on a mass of one kilogram giving it a velocity of one meter per second per second.

One Newton =  $1 \text{ kg.m/s}^2 = 0.2248 \text{ lb. force}$ .

See **Force**.

**Newtonian fluids** Sir Isaac Newton (1642–1727) a British mathematician and physicist, was instrumental in the study of fluid mechanics and when he published his *Principia* in 1687, he stated that both water and air act in accordance with Newtonian fluid flow. Regardless of any force acting on the air or water, they continue to flow as a uniform stream of particles.

Newtonian fluids also depend on the constant of proportionality, known as the coefficient of viscosity. The viscosity of air can be considered to increase with a temperature rise but is virtually unaffected by the air pressure, as opposed to a liquid fluid where the temperature and pressure must both be considered. Newtonian fluids is also known as Newtonian mechanics.

See **Viscosity**.

**Newtonian theory of lift** Newtonian physics is the basis for the Newtonian theory of lift, with reference to Newton's laws of motion. According to Newton's third law of motion, the lift produced is equal and opposite to the rate of change in the air momentum as represented by the downwash vector. Greater lift can be induced either by increasing the angle of attack and hence, diverting a greater air mass downward, or by increasing the airplane's forward velocity; either will increase the vertical component of the downwash, and increase the lift. It follows:

- the volume of diverted air is proportional to the ambient air density and the flow velocity
- the vertical velocity of the downwash flow is proportional to the angle of attack and the flow velocity
- lift is proportional to the vertical velocity of the downwash times the mass of air in the downwash. (Newton's 2<sup>nd</sup> law where force =  $ma$ )
- The wing induces downwash and the equal and opposite reaction is for the airflow to push the wing upwards to create lift. (Newton's 3<sup>rd</sup> law of equal and opposite motion).

This explanation of lift is good as far as it goes. However, if there is a vertical component to the downwash vector, then there must also be a horizontal component involved. Applying Newton's 3<sup>rd</sup> law, the force equal and opposite to the horizontal, rearward facing component must be a horizontal force acting in a forwards direction! This would be a thrust force producing eternal motion, which is impossible. The horizontal component is never considered in the Newtonian theory!

See **Downwash**.

**Newton, Sir I.** Sir Isaac Newton (1642–1727) a British mathematician and physicist, devised three now well-known laws of motion (straight and circular) that describe the relationship between forces and the effects a force has on a body. Knowledge of these three laws (introduced in 1686) is essential for a clear understanding of the basics of aircraft flight. These laws are applicable to many situations in physics and also of course, aerodynamics and meteorology, etc. Briefly, Newton's laws are:

- first law – the law of inertia
- second law – the conservation of momentum,  $F = ma$
- third law – the law of action and reaction.

He was also the first person to make early calculations of the speed of sound, which were later corrected by Pierre-Simon



Laplace (1749–1827) when he allowed for the action of the gas molecules.

See **Newton's First Law of Motion; Newton's Second Law of Motion; Newton's Third Law of motion.**

**Newton's First Law of Motion** Newton's First Law of Motion states, 'A body (aircraft) has a tendency to continue doing what it is doing. When at rest it continues to remain at rest and when in motion it remains in motion at the same speed and in the same direction, unless acted upon by an outside force.' This law is also known as the law of inertia. A stationary aircraft on the ramp will stay there and an aircraft in straight and level flight will continue its flight unless disturbed. The law describes the influence acting on the body as a force; the force affects the equilibrium of the body.

Newton's First Law of Motion can also be applied to torque, rephrased, as 'a rotating body will continue to rotate in the same direction and at the same speed until an outside force acts upon it'.

**Newton's Second Law of Motion** 'The rate of change in motion or momentum is directly proportional to the size of the force, or inversely proportional to the mass of the body (aircraft). The rate of change in motion will be in the direction of the applied force'.

This second law, the conservation of linear momentum, relates the description of motion to its cause and relates the effects of a force to acceleration defined by:

$$F = ma$$

Where  $F$  = force

$m$  = mass

$a$  = acceleration.

The second law when related to curvilinear motion is stated as:

$$F = mv^2/r$$

Where  $F$  = force

$m$  = mass

$v^2$  = velocity squared

$r$  = radius.

The second law also describes how a body responds to a force, thus connecting force with motion and acceleration. To make an aircraft turn requires a force in the direction of the turn. The greater the force, the greater the rate of movement. Centripetal and centrifugal forces are good examples of forces in circular motion. It should be noted here that the forces are vector quantities and therefore both direction and magnitude are as important.

**Newton's Third Law of motion** Newton's Third law of Motion states, 'for every action, there is an equal and opposite reaction'. The law states how an action and reaction pair of forces is related; this is important in the study of energy and momentum. Understanding the third law is essential in our study of aircraft propulsion, where the action of thrust produces an equal and opposite reaction to propel the aircraft through the air. The lift force acting on an aircraft in straight and level flight give a good example of equal and opposite reaction where lift equals weight and thrust equals drag.

The third law applies only when there is no acceleration involved.

**Nibble** Nibble is an alternate name for burble, which occurs during high angles of attack when the airflow over the main wing breaks away as a turbulent wake and flows over the tailplane. This is felt by the pilot as vibration, or buffeting through the control yoke.

See **Burble/burbling; Separation point & flow; Stagnation point/line.**

**Nominal pitch** The nominal (geometric) pitch of a propeller is stated for a nominated propeller radius, usually at the 80% station.

It is also known as standard pitch.

**Non-differential spoilers** All spoilers across both main wings extend a like amount to act as airbrakes with or without roll control being applied.

**Non-lifting tailplane** In normal cruise flight the tailplane will experience a down load, in effect – upside-down lift. This is seen as extra weight, which the main wings must counteract. Drag is also produced.

A non-lifting tailplane is of symmetrical design and is placed at zero degree angle of attack during cruise flight. Therefore, being a symmetrical airfoil streamlined to the airflow, there is no unwanted down force during the cruise and drag is at a minimum.

**Non-planar wings** A planar wing consists of a straight wing with either dihedral or anhedral as opposed to a non-planar wing, which has its dihedral angle increased somewhere along the span, usually at about the quarter to half span location. A good example of a non-planar wing is found on the Fletcher FU-24 top dressing



**Photo:** The New Zealand built Fletcher Cresco, turboprop ag-plane has non-planar wings.

aircraft, the McDonnell F4 Phantom II and a few other aircraft. Gull wings are also a type of non-planar wing. An alternative, but less common name is a polyhedral wing.

*Compare with* **Gull wing; Inverted gull wing.**

**Normal axis** The OZ vertical or normal axis is considered a straight line through the centre of gravity at right angle to the lateral and longitudinal axes.

*See* **Diagram 2, Planes of Movement; Axes; Aerodynamic axes.**

**Normal category** All aircraft are included in the normal category and some also fall into the utility and/or aerobatic categories. The normal category allows all normal manoeuvres to be flown, including stalls (but not whipstalls) and steep turns not exceeding 60 degrees angle of bank.

The American FAA Part 23 normal category limits are positive 3.8g and minus 1.52g. With flaps down, the load factor may be reduced on some aircraft.

FAA Part 23 was introduced in 1965 and covers all light aircraft up to 12500 pounds MAUW and commuter aircraft up to 19000 pounds MAUW. FAA Part 25 covers all transport aircraft above 19000 pounds. All aircraft with a MAUW above 300 000 pounds are classed colloquially, as Jumbo Jets or 'heavies'.

*See* **Limit & ultimate load factors; Ultimate load factor.**

**Normal operating speed ( $V_{NO}$ )** The normal operating speed is marked at the top end of the green arc on the air speed indicator. The  $V_{NO}$  is the same as cruise speed ( $V_C$ ).

*See* **Manoeuvre envelope; Gust envelope.**

**Normal pressure drag** The normal pressure drag ( $C_{DP}$ ) includes the wave drag, induced drag and form drag all combined as one drag type.

*See Drag.*

**Normal propeller state** A normal propeller state is achieved when the propeller is producing power and the thrust is acting forwards in the direction of flight.

*See Propeller forces in cruise flight.*

**Normal shockwave** A normal shockwave stands at  $90^\circ$  (normal or perpendicular) to the relative airflow at about mid-chord on the wing's upper surface at a speed of Mach one. It has a subsonic flow behind the shockwave. With increasing Mach number, the normal shockwave leans further rearward to become an oblique shockwave, weakening as it does so.

The Frenchman, Pierre Hugoniot discovered the normal shock wave in 1887.

Alternate names are weak or incipient shockwaves.

**Normal spin** A normal spin is one entered (usually) by pilot input from controlled straight and level flight with normal recovery action stopping the spin within two turns.

*See Inverted spin; Precision spin; Spinning.*

**Northrop, J.** John (Jack) Knudsen Northrop (1895–1981) was an American industrialist and in 1927 became a co-founder of the Lockheed Corporation. In 1935, he established his own company – the Northrop Corporation building several well-known aircraft. He also specialized in flying wing aircraft and the Northrop N-1M, N-9M, YB-35 and YB-41 emerged from his stable. For his contribution to American aviation, he was awarded in 1947, the Medal of St. Louis from the American Society of Mechanical Engineers.



**Photo:** The Northrop N-1M single-seat, flying wing was one of Northrop's early designs.

**Nose up/down** This is a pilot's reference to the pitch attitude of the aircraft as seen from the pilot's seat as the front of the aircraft pitches up or down.

**Nose plane** Some of the very early type of aircraft had their stabilizer wing mounted on the forward part of the fuselage near the 'nose'. The modern canard wing is not considered as being a nose plane.

**No-slip condition** Consider a laminar boundary layer in a viscous flow passing over a wing. The molecules close to the surface may become stationary with molecules above gradually increasing speed up to the top of the boundary layer. Where the molecules become stationary on the surface due to the shear stress and velocity gradient of the viscous flow, it is known as the 'no-slip condition'.

*See* **Boundary layer.**

**NOTAR** The McDonald Douglas helicopter manufacturer coined the term NOTAR (the acronym for no tail rotor) when they introduced their new McDonald Douglas 520N NOTAR helicopter. In place of the tail rotor, compressed air is forced out of slots on the right-hand side of the tail boom and combined with the downwash from the main rotor, the resulting Coanda effect is used to counteract main rotor torque. Additional yaw control around the OZ vertical axis is achieved by varying the amount of thrust produced from the jet thruster unit mounted at the end of the tail boom.

In addition to the above, a strake is mounted on the left-hand side of the tail boom to spoil the airflow on that side of the boom to accentuate the Coanda effect of the ejected airflow on the right side. Twin vertical tails also assist in the directional stability when the helicopter is flying with forward speed.

The NOTAR system is far quieter and safer for nearby ground crew than a conventional anti-torque rotor system.

The MD 520N NOTAR made its first flight on 28 December 1989 followed shortly after by the more powerful MD 530N. The MD 520 helicopter originated from the Hughes 500 series. However, the NOTAR concept was first introduced and tested on a Hughes OH-6A helicopter in 1981.

*See* **Coanda effect.**

**Notch** A notch is a chordwise saw-cut on the leading edge of an airfoil. It is distinguished by the inner span having a distinct chord less than that of the outer span.

On sweptback wings, a notch helps prevent flow separation and stalling near the wingtips to reduce large pitching moment changes. Also known as, a dog tooth or saw tooth.

*See* **Rebated leading edge; Leading edge devices (LEX).**



**Photo:** The Hawker Hunter is one of many types of aircraft with a leading edge notch. This Hunter is located in the Classic Flyers Aviation Museum in Tauranga, New Zealand.



**N-wave** The N-wave is associated with the pressure signature of sonic booms from aircraft flying at supersonic speed. It comprises the effects of compressibility in a sonic compression wave (shockwave). The N-wave is also present during the formation of a Prandtl-Glauert singularity on fighter aircraft as they traverse the transonic region.

The profile of the shockwave is shaped like the capital letter 'N' when plotted on a graph. The forward shockwave (measured in pounds per square foot against time in milliseconds) can be considered as the first vertical stroke of the 'N' and the rear shockwave as the second vertical stroke. The first shockwave produces an increase in positive air pressure – an overpressure – or leading compression through the shockwave. The peak overpressure is in the range of 1 to 10 pounds per square feet. After the first shockwave, there exists a linear decrease in pressure, a rarefaction or decrease in air density with the air velocity increasing, represented by the diagonal down-stroke of the 'N'. Following the tail end shockwave, the air pressure returns to normal ambient conditions, represented by the last vertical stroke of the 'N'.

Steady straight and level flight at supersonic speed produces an N-wave as opposed to a U-wave (a focused boom) which is generated in supersonic manoeuvring flight.

The N-wave phenomenon was discovered during flight-testing of the North American B-70 Valkyrie supersonic bomber.

It is also known as the pressure signature.

*Compare with* **U-wave**. See **Prandtl-Glauert singularity**.

# O

**Oblique shockwave** An oblique shockwave is formed in a supersonic compression flow when the airflow has exceeded Mach 1.0, or when the airflow is forced to change direction at a concave corner such as in front of a cockpit canopy. The oblique shockwave is formed at the base of the wedge (canopy) and therefore, the airflow changes direction due to the shockwave – not the canopy. It can also occur at the leading edge of the body – aircraft wing, nose or a contracting duct inlet or any time a compression flow is formed. With increasing speed the oblique shockwave arcs further rearwards decreasing the wave angle  $\beta$ . The angle made between the wedge and the upstream flow is the deflection angle  $\theta$ .

The velocity through the shockwave decreases as the density, temperature, and pressure increases in accordance with the conservation of momentum, mass and energy as deduced from the Rankine-Hugoniot equation. Although in general, the airflow behind the oblique shockwave is nearly always supersonic.

Ernst Mach (1838–1916) was the first to discover the oblique shockwave phenomenon when he was studying ballistics. During the time from 1905 to 1908, Professor Ludwig Prandtl (1880–1953) and his student Theodore Meyer at Gottingen University, carried out further investigation into the physics of the oblique shockwave.

*See Diagram 57, Mach Angle & Cone; Diagram 15, Compression Flow & Oblique Shockwave; Rankine-Hugoniot equation; Shockwaves.*

**Oblique wing** An oblique wing is one designed to pivot on top of the fuselage. For low-speed flight, the wing sits at right angles to the fuselage. As speed increases, one wing sweeps forward, the other aft, in unison, as opposed to a swing-wing aircraft where both wings sweep aft simultaneously. At Mach 1.4, the wing sweep needed is about  $60^\circ$  and at Mach 2.0, the required sweep is about  $70^\circ$ .

The oblique wing idea has a few advantages over a normal swept back (or forward sweep wing) as follows:

- a low-speed airfoil when it is un-swung
- swept for high-speed flight
- only one pivot is required, which reduces bending loads compared to two pivots on swing wing aircraft
- weight is reduced due to less structure and complexity of a one piece wing
- pivot actuator loads are reduced
- trim changes are small
- the aircraft is more stable than conventional swing wing aircraft
- supersonic wave drag is reduced
- The aerodynamic centre location is relatively constant.

A proposed variation on the oblique wing concept is an aircraft with no fuselage. It is in effect, a flying wing with the sweep angle adjusted to the line of flight by swiveling the jet engines and vertical fins.

The idea, introduced by German designer Dr. Richard Vogt (1894–1979) of the Blohm & Voss Company found its way to the NASA AD-1 research aircraft, which was designed jointly by NASA's Ames and Dryden Research Centres, to test the concept of the oblique wing aircraft for supersonic flight. The chief designer of the AD-1 was Robert T. Jones (1910–1999) at the Dryden facility, with wind tunnel testing performed at the Ames facility. The AD-1 flew its first test-flight on 21 December 1979. The AD-1 is a single-seat jet-powered aircraft with a rigid wing pivot-mounted on top of the fuselage and was used primarily for low-speed flight research.

The oblique wing is also known as a slew wing, skew wing or scissor wing.

**Offset fin** The fin/rudder assembly of a single-engine light aircraft may be offset at an angle to the aircraft's centreline or OX longitudinal axis to counteract effects of the propeller slipstream. Usually it is offset a pre-determined amount to provide sufficient counteracting force in cruise flight at normal cruise speed and cruise power settings. At other speeds and power settings rudder trim adjustments will be required inputs by the pilot.

**Offset flapping hinge** Offset flapping hinges are designed to reduce the swinging of the fuselage beneath the helicopter's rotor disc to improve lateral and longitudinal stability.

In addition, if the centre of gravity is near its fore or aft limits on lift-off to the hover, the fuselage may hang down out of alignment with the rotor disc, which is an undesirable condition. Increased control power (reduced swinging of the fuselage under the rotor disc) assists the fuselage to remain more or less parallel to the rotor disc during manoeuvring while the centre of gravity range is extended and the maximum forward speed is increased. It is achieved by mounting the flapping hinge a short distance outboard from the masthead. It then produces a turning couple when the rotor blades flap up and down in opposition as they rotate. Assisted by the rotor blades centrifugal force, the mast and hence the fuselage is brought into alignment with the rotor disc overcoming the pendulosity forces.

*See Pendulosity; Control power.*

**Offset pitch horns** Offset pitch horns provide a means to reduce the flapping experienced by the helicopter's rotor blades.

The offset pitch horns are attached to the rotor blade's leading edge, such that when the rotor blade flaps up, the horn restricts the ability of the leading edge of the blade to rise. Therefore, as the blade rises, the blade angle reduces; the opposite action occurs when the blade descends. The amount of lift coefficient increase or decrease is unaffected by flapping.

*See Flapping.*

**Ogival/Ogee delta wing** The Ogival (Gothic) delta wing is a modified low aspect ratio planform of a basic delta wing resembling a Gothic church window (or a wine glass). That is, one with the leading edge planform radius increasing to become parallel to the aircraft's centre line – a cropped compound delta.

The planform of Concorde's delta wing has a varying curved leading edge, its planform being known as an ogee wing, a variation on the ogival planform. The advantage of ogival/ogee (slender delta) wings is that the entire wing lies within the Mach cone at supersonic speeds of around Mach 2.0. Around one hundred different wing shapes were tested for the Concorde.

It is also known as a slender delta wing.



**Photo:** The Concorde's wing is a beautifully shaped ogive wing planform. This Concorde is in the Fleet Air Arm Museum in Yeovilton, England.

**Onglet** An onglet is a fillet located on the wing root leading edge.

See **Fillets**.

**Open circuit wind tunnel** An open circuit wind tunnel has each end open to atmosphere and is devoid of a return flow circuit. At the upwind end, the motor and fan are located while at the inlet end vanes are used to straighten out the airflow as it enters the tunnel.

*See* **Wind tunnels.**

**Open cycle** The open cycle is a continuous thermodynamic process and is commonly associated with the gas turbine (jet) engine.

The airflow through the gas turbine engine's four stages of compression, combustion, expansion, and exhaust consists of two adiabatic and two isobaric, alternating changes.

*See* **Brayton cycle; Adiabatic process; Isobaric.**

**Open jet wind tunnel** An open jet wind tunnel has an open area for access to the plane or model while the wind tunnel is in operation, as opposed to a closed jet wind tunnel where access is not available to the plane or model. An open jet tunnel may, or may not be an open circuit tunnel.

*See* **Wind tunnels.**

**Operating envelope** An alternate name for the V-n manoeuvre envelope is the operating envelope.

*See* **Manoeuvre envelope.**

**Optimum cruise altitude** At the optimum cruising altitude for a piston-powered aircraft, the indicated air speed (IAS) is lower than the true air speed and is closer to the best range speed, where the requirement for the least amount of fuel used per unit distance will be achieved. The higher true air speed (TAS) for the same fuel flow at the optimum cruise speed satisfies the second requirement where the least amount of fuel used per unit velocity is achieved. By flying higher, the true air speed increases for a given indicated air speed, or in other words, the IAS/TAS ratio increases with altitude. Therefore, in regards to the IAS/TAS ratio, flying at higher altitudes improves the piston-powered airplane's efficiency.

**Optimum cruise speed** The aircraft's cruise speed at which the least amount of fuel is used per unit velocity is known as the optimum cruise speed.

**Optimum glide speed** The optimum glide speed determines the best speed to use to allow for the variation in rate of sink in head or tail winds. Pilots of sailplanes use the optimum glide speed to extract more performance from their craft.

**Orthogonal biplane** The early biplanes had each set of wings mounted directly one above the other without any stagger; they were known as orthogonal biplanes. Later biplanes had their wings staggered out of alignment to reduce the effects of biplane interference. Usually the top wing has forward stagger, as found on the DeHavilland DH82 Tiger Moth, as opposed to the Beech Model 17 Staggerwing with negative, back stagger.

See **Stagger; Biplane interference; Negative stagger.**



**Photo:** The Spad VII is a WWI fighter with an orthogonal biplane configuration. This Spad is located at the National Museum of the USAF in Dayton, Ohio.

**Oswald efficiency factor** The Oswald efficiency factor ' $e$ ' is a correction factor relating the change in drag coefficient due to a change in lift coefficient. It was introduced by, and named after Dr. W. B. Oswald, who conceived the idea.

The aircraft's total variation of drag with lift is expressed by the factor of ' $e_o$ ', which would theoretically be a value of one for a complete aircraft with a perfect elliptically shaped wing and an unhindered wing upwash. [The factor for a set of wings detached from the aircraft is ' $e$ ']. However, in practice, the value is always less than one and for most aircraft, the value lies between 0.7 and 0.9. An aircraft with a high aspect ratio wing would have a relatively high efficiency factor, as opposed to an aircraft with a low aspect ratio wing as commonly used on supersonic wings. An Oswald efficiency factor between 0.3 and 0.5 would be average values at



Mach 1.2. Therefore, higher values of the Oswald efficiency factor are found on more aerodynamically efficient aircraft.

Some sources quote a lower case 'e' as the 'span e' when referring to the wings only, and a capital 'E' is used for the whole aircraft and is known as 'Oswald's E'.

The Oswald efficiency factor is found from the following formula:

$$C_D + C_{D0} = \frac{(C_L)^2}{\pi e_o AR}$$

Where  $C_D$  = overall drag coefficient

$C_{D0}$  = zero-lift drag coefficient

$\pi$  = 3.14

'Span e' or e = Oswald efficiency factor for wing only

Oswald's E or  $e_o$  = Oswald efficiency factor for whole aircraft

AR = aspect ratio.

It is also known as the span efficiency factor.

**Oswatitsch inlet** *See* Centre body cone inlet.

**Outflow velocity** The propeller under normal operating conditions sucks air into the front of the propeller disc increasing its pressure and ejects it behind the propeller as slipstream/thrust. As it does so, the slipstream's outflow velocity continues to accelerate reaching its maximum outflow velocity some distance behind the propeller.

*Compare with* Inflow velocity.

**Out of ground effect (OGE)** A helicopter is out of ground effect when the helicopter is hovering more than one rotor disc diameter above the ground.

*Compare with* In ground effect (IGE).

**Outside wing** The wing furthest from the centre of the turn is the outside wing. Its air speed will be greater than the inside wing's speed due to traveling a greater distance and will produce more lift causing the aircraft to over bank.

*See Inside wing.*

**Overhang** Overhang refers to the wingspan of biplane wings, where the upper wing is of a greater span than the lower wing. It is considered positive when the upper wing is the longer of the two, the most common lay out. The purpose of overhang is to ensure adequate lower wingtip clearance during ground operations; the lower wing may additionally have dihedral for the same purpose. Overhang is measured along the span of the upper wing to a point vertically above the tip of the lower wing; it is then calculated as a ratio from the following formula:

$$\text{Overhang} = \frac{100 \times \text{upper span} - \text{lower span}}{\text{lower span}}$$

The overhang on a monoplane is the spanwise distance from the wingtip to the strut.

*See Sesquiplane.*

**Over pitching** Over-pitching of a helicopter's rotor blades is due to incorrect pilot input, particularly when approaching the hover. An over-pitch causes the rotor blade's angle of attack to be too high. This causes a decrease in rotor RPM and a decrease in total rotor thrust (TRT). Increased up-collective by the pilot, causes an increase in the coning angle, which tilts the rotor thrust inwards, decreasing the TRT/rotor drag ratio further, which engine power may not be able to overcome causing a decrease in rotor RPM and a rapid increase in helicopter sink rate.

**Overpressure** A supersonic aircraft produces shockwaves, which are heard as sonic booms by observers on the ground below the flight path. The sonic booms are caused by an increase in air pressure; this is known as the overpressure.

*See* **Sonic boom; Blunt nose re-entry vehicle; Refracted boom.**

**Overspeed Condition** A constant-speed unit malfunction may cause the propeller blades to move into full-fine pitch in flight and overspeed, or run-away. The engine RPM may rapidly increase and exceed the maximum limits, which may be accompanied by a high pitch whining noise caused by the very high, propeller tip speed. The engine should be throttled back and shut down immediately to prevent damage.

# P

**Padding** *See* Ground resonance.

**Paddle blade** A paddle blade is a wide chord, low aspect ratio propeller blade. A paddle blade has greater solidity and a higher activity factor and will absorb more engine power than a high aspect ratio propeller blade. However, its efficiency may be reduced due to the blade wake affecting the thrust produced by the previous blades.



**Photo:** The Lockheed C-3 Orion anti-submarine patrol aircraft has four Allison 4910 ESHP T-56 turboprop engines each driving four paddle-blade propellers. Note the blade root cuffs for increased propeller efficiency.

**Pancake** Modern twin-engine jet fighters with a variable-geometry wing have a large flat area on top of the fuselage known as the pancake. It covers the area above the swing-wing mechanism; it provides some lift and it may also house a fuel tank.

It is also known as a glove.

**Parallelogram of forces** The parallelogram of forces is a vector diagram showing the magnitude and direction of a given force divided into two or more components, not forming any right angles. Alternatively, it can be shown as the resultant of two different forces represented by the sides of a parallelogram with the resultant force drawn diagonally from their origin and bisecting the drawn sides. These diagrams are frequently found in aerodynamic text.

**Parasite drag** Parasite drag is the aircraft's total drag minus the induced drag. It includes profile drag (form drag), skin friction drag, rotor blade drag plus body drag and extra drag, in fact all non lift-dependant drag, although it is related indirectly to lift and varies directly with dynamic pressure and area. It also increases exponentially, and varies as the speed squared – double the speed and parasite drag increases four times. Gear and flap deployment also increase parasite drag.

Parasite drag is also known as the zero lift drag, and is related to 3-D aerodynamics.

*See Drag.*

**Parasite drag coefficient** A dimensionless coefficient or constant multiplier. The aerodynamic cleanness of the aircraft is measured by the parasite drag coefficient. The amount of dynamic pressure ( $\frac{1}{2} \rho V^2$ ) converted to drag is expressed through the parasite drag coefficient.

$$\text{The parasite drag coefficient} = C_{DP} = \frac{\text{Parasite drag}}{\frac{1}{2} \rho V^2 S}$$

Where  $C_{DP}$  = parasite drag coefficient

$\frac{1}{2}$  = a constant

$\rho$  = air density kg/m<sup>3</sup> or lbs/cu.ft.

$V^2$  = aircraft speed in m/sec<sup>2</sup> or FPS<sup>2</sup>

$S$  = wing area in square meters or feet.

*See Drag.*

**Parasite power** Parasite power varies as speed<sup>3</sup>. Due to parasite drag increasing with the forward speed squared ( $V^2$ ), the parasite power must also increase to equal it, starting from zero power at zero forward speed.

See **Power curves – helicopter**.

**Parasol wing** A high wing supporting the fuselage below on a pylon is known as a parasol wing.



**Photo:** The Consolidated Catalina flying boat must be one of the largest planes with a parasol wing. This plane is an OA-10 version of the PBY Catalina and is displayed at the RAF Cosford Museum, England.

**Partial flap** The use of partial flap, the first 5–15 degrees of flap lowered is used for take-off and the initial approach to land. It is also known as the take-off, or optimum flap.

**Pascal** The Pascal is the metric unit of pressure, where the Pascal, Pa = N/m<sup>2</sup> .

*See Pressure.*

**Path length** The path length compares the distance the airflow travels over the upper and lower wing surfaces from the leading edge to the trailing edge. A wing with greater camber or one with area increasing flaps and/or leading edge devices extended can increase the path length with the upper surface path length always being longer than the lower surface. The difference in path length of the upper and lower surfaces only partly contributes to the generation of lift.

**Pathline** In an unsteady turbulent flow over a body, the air particles flow in a random manner. The path taken by the individual particles is drawn on a diagram as the pathlines. Pathlines are the equivalent of a laminar flow's steady streamline where all the air particles are flowing the same way.

**Peak suction** Peak suction is the area on the upper surface of an airfoil where the air pressure is at its lowest pressure.

*See Diagram 53, Lift Distribution.*

**Peaky** A lift coefficient curve is said to be peaky when the top of the curve at the maximum lift coefficient, drops sharply with a small radius curve. A peaky curve is present when the large peak suction is relatively close to the wing's leading edge – around 10–15% MAC and the wing stalls abruptly and sharply. This is in contrast to a supercritical airfoil where the top of the lift curve is a more gentle-curving arc (a roof top curve) with a progressive stall break with the suction area located further aft on the wing.

*See Diagram 52, Lift Coefficient V. Straight, Swept and Delta Wings.*

**Pendulosity** The swinging pendulum motion of an airplane particularly in turbulence. It is also inherent in helicopters where the fuselage may tend to swing underneath the rotor disc due to turbulence, manoeuvres or with the centre of gravity near its fore and aft limits of travel.

*See* **Control power.**

**Pendulum mode** *See* **Air resonance.**

**Pendulum stability** Pendulum stability is inherent on high-wing aircraft where a relatively large vertical distance (and moment arm) exists between the centre of lift and the aircraft's centre of gravity; the centre of gravity is located vertically below the lift. Therefore, when these two forces are out of alignment – which would be the case if the aircraft were rolled by turbulence – and producing a large moment about the centre of gravity, the aircraft would sideslip and the two forces of lift and weight would produce a restoring moment to bring the aircraft back to wings level flight. Pendulum stability is reduced on low wing airplanes due to less vertical distance between the centre of lift and centre of gravity; hence the need for dihedral.

Pendulum stability is also known as the Keel effect.

*See* **Dihedral angle; Dihedral action/effect.**

**P-factor** The term P-factor is related to the cause of swing on a tail-wheel aircraft during take-off.

The term P-factor does nothing to describe the cause of the swing, however the full term of propeller factor and the alternate names of asymmetric blade effect or asymmetric disc loading are more descriptive. The asymmetric blade effect is associated with the propeller's blade element theory and describes the difference in thrust on the up and down-going blades. The asymmetric disc loading is associated to the axial momentum theory, which deals with the air mass flowing through the propeller disc.



Whenever P-factor is present, one half of the propeller disc produces more thrust than the other half, due to the propeller axis being inclined to the direction of flight; it is more prominent on tail-dragger aircraft during take-off. This is due to the fact propeller thrust is greatest at low-speeds (see *Diagram 60, Thrust & THP Required & Available Curves*). To describe P-factor we will assume the propeller has two blades. P-factor is normally attributed to the difference in angle of attack between the up-going and down-going blades. This is true to some extent but the difference in velocity between the two blades must also be taken into consideration, because velocity is the major factor.

When the propeller axis is parallel to the airflow through the propeller disc, the angle of attack of both propeller blades remains the same. As the angle of inclination of the propeller axis is increased, the difference in angle of attack also increases up to a maximum difference at 45° inclination. From 45° inclination on up to 90°, the difference in angle of attack reduces back to zero degrees. Consider the angle of propeller axis inclination on a tail-dragger on the ground. The difference in angle of attack is relatively small at approximately ½° to 1°, producing a difference in lift coefficient of about 6%. The angle of attack on the down-going blade is the greater, producing more thrust on that side of the propeller axis. However, due to the inclined propeller axis the up and down-going blades meet the airflow with a difference in velocity of around 7%.

It must now be remembered from the thrust (lift) formula that velocity is squared. Therefore, the effect of velocity is much greater than that of the angle of attack. With the propeller axis inclined, the speed ratio of both blades is different. At a constant forward speed and constant RPM, this may at first sound puzzling. So, let us take the extreme case with the propeller axis inclined 90° to the direction of airflow, as found more or less, on a helicopter. The advancing rotor blade has its own rotational speed plus the forward speed of the helicopter. The retreating blade also has its own rotational speed minus the helicopter's speed. This produces

a difference in speed between the two blades. The inclined propeller experiences the same effect to a much lesser degree, but the difference in speed is still there. The down-going propeller blade is akin to the helicopter's advancing rotor blade. Having the greater velocity and angle of attack it produces the maximum thrust on that half of the propeller disc. The blade root area has lower rotational velocity than the outer portion of the blade where the difference in angle of attack has greater influence in producing thrust than the difference in blade speed. On the outer portion of the blade where thrust is greater, rotational velocity has greater influence over the angle of attack.

The result is, P-factor is a combination of both the angle of attack and rotational velocity being greater on the down-going blade and with rotational velocity being the major contributor to asymmetric disc loading or P-factor.

*See **Asymmetric disc loading; Asymmetric thrust.***



**Photo:** The Republic P-47 Thunderbolt's 2300 horsepower engine driving a four-blade propeller is a prime candidate for P-factor. This P-47 resides at the National Museum of the US Air Force in Dayton, Ohio.

**Phase lag** The helicopter's rotor blade phase lag refers to the delay in the rotor blade reaching its highest (or lowest) position as it rotates around the disc. In order to tilt the rotor disc to perform a desired manoeuvre, the rotor blade will start to rise (or descend on the opposite side of the disc) approximately  $90^\circ$  before and after its required position respectively.

The amount of phase lag (from  $80^\circ$ – $90^\circ$ ) depends on the location of the flapping hinges with respect to their distance from the rotor mast. The greater the distance of offset from the mast, the less the phase lag.

**Phillips, H.** The Englishman Horatio Frederick Phillips (1845–1924) introduced some major contributions to aerodynamics.

He conducted some early and significant wind tunnel tests and patented the first cambered airfoil designs in 1884; it was George Cayley (1773–1857) who was the first to propose using the cambered surface. Horatio Phillips noted in his studies, airfoils with upper and lower surfaces, as we know them today, are far more efficient than the early thin airfoils. He also discovered the suction on the upper surface of airfoils combined with a thicker leading edge, provided the greater part of the lift.

Horatio Phillips also designed and built early aircraft with multiplane wings. Looking very much like flying Venetian blinds, one of his planes had 200 of these panels but none of his designs was successful.

**Phugoid motion** Following a disturbance from a trimmed and level flight attitude, the aircraft enters a long-term, low frequency oscillation (phugoid motion) in the pitching plane around the OY lateral axis. Positive longitudinal stability will quickly damp out the phugoid motion. Neutral or negative longitudinal stability will allow the motion to continue, or diverge respectively. The length of time for the aircraft to progress through a phugoid motion is determined by the aircraft's trimmed air speed; its altitude and flying characteristics have no effect.

Dr. Frederick William Lanchester (1868–1946) studied and introduced the term phugoid motion to aerodynamics in 1908.

**Pilot assisted** See Pilot induced oscillation (PIO).

**Pilot induced oscillation (PIO)** An airplane's short period longitudinal motion around the OY lateral axis coupled with the pilot's input frequency can cause a system response-lag leading to pilot induced oscillations.

The oscillations can be caused by the pilot, the plane or both, especially pilots inexperienced on the aircraft type; for example student pilots may have a problem adjusting the pitch attitude to flare the aircraft for landing and over-pitch (or balloon) requiring a further correction, which leads to PIO.

Low stick force per g may be present during high-speed and low altitude flight, which combined with a pilot's out of phase control input can lead to oscillations. PIO is more inherent in short-coupled aircraft – aircraft with a relatively short distance between its main wing and the tailplane. The resulting unstable oscillation can cause airframe overstress or failure if the oscillations become extreme. The cure to reduce the pilot induced oscillations is for the pilot to release the flight control allowing the aircraft to recover through its own inherent stability.

Pilot induced oscillation is also known as aircraft-pilot coupling or pilot assisted, or augmented oscillations.

**Piston-engine power curves – airplane** The piston-engine power curve graphs, represent an individual aircraft's performance. From these graphs, we can find the maximum and minimum flight speeds, the rate of climb or descent and the range and endurance speeds, etc. With reference to *Diagram 60, Thrust & THP Required & Available Curves*, the horizontal line at the top of the graph represents the engine's brake horsepower and it will descend the graph (remaining horizontal) as altitude is increased or the engine is throttled back. Below this line are the two curved lines for thrust horsepower available and thrust horsepower required. These two curves intersect twice – at the minimum speed ( $V_{S1}$ ) and at the maximum cruise speed ( $V_{CR \text{ max}}$ ). The power or thrust required

is determined by the aerodynamics of the aircraft as opposed to the power or thrust available, which is determined by the engine's output.

A dotted line shows the lower end of the thrust horsepower available. This is due to Drzewieck's fallacy where the thrust horsepower equals drag times velocity. If the aircraft is stationary, the speed is zero (and the drag is zero) because any number multiplied by zero gives a zero answer. Therefore, according to this rule, no thrust horsepower is developed and the aircraft would not move. However, reference to **Diagram 60A**, shows that the static thrust is at a maximum and therefore the aircraft will and does move from a stationary position. The distance between the brake horsepower curve and the thrust horsepower available curve represents the power available from the engine to the propeller.

A tangent drawn from the origin (0) intersects the thrust horsepower required curve. A vertical line drawn down from the intersection of the tangent and the thrust horsepower required gives the TAS range speed (it is also the  $V_Y$  minimum drag speed) although the actual cruise speed used in practice is 1.32 times  $V_Y$  range speed. Both of the lines representing the brake horsepower and thrust horsepower available descend the graph with reduced power – because of either an increase in altitude or a decrease in power.

The thrust horsepower required curve moves up the graph and to the right, when plotted for true air speed whenever the aircraft climbs to altitude. When these two curves meet, the aircraft will be at its absolute ceiling. Reducing engine power by closing the throttle will bring the thrust horsepower available curve down the graph. When it reaches the thrust horsepower required curve, the rate of climb will be zero and the aircraft will only just maintain level flight; this is the situation when the engine is throttled back to cruise power. The thrust horsepower required curve is similar to the total drag curve, where the thrust increases above and below the minimum drag speed, after all, thrust equals drag at all speeds. It should be remembered that because drag increases with

the square of the speed, it follows that power-required increases with cube of the speed.

The difference between the thrust horsepower required and thrust horsepower available is the power differential (Pd) or excess thrust horsepower where the  $V_Y$  maximum rate of climb speed is achieved. A horizontal line intercepting the bottom of the thrust horsepower required equals the  $V_X$  maximum angle of climb speed. This is also the minimum power speed and endurance speed.

The lower graph represents the thrust-pounds required and thrust-pounds available. It can be seen the thrust differential (Td) is the maximum difference in thrust available and thrust required which occurs at the  $V_X$  minimum drag speed. The intersection of the thrust-pounds required and available curves coincides with the maximum cruise speed. A decrease in power will bring the thrust-pounds available curve down the graph, which in turn will move the intersection of the thrust-pounds available and required towards the left in line with decreasing maximum cruise speed. [The top graph is drawn in terms of horsepower and the lower graph in terms of pounds of thrust]. The formula below shows the thrust horsepower required is related to the thrust-pounds required and the aircraft's desired speed.

$$\text{Power} = \frac{\text{TrV}}{325}$$

Where Power = thrust horsepower required

Tr = thrust required in pounds

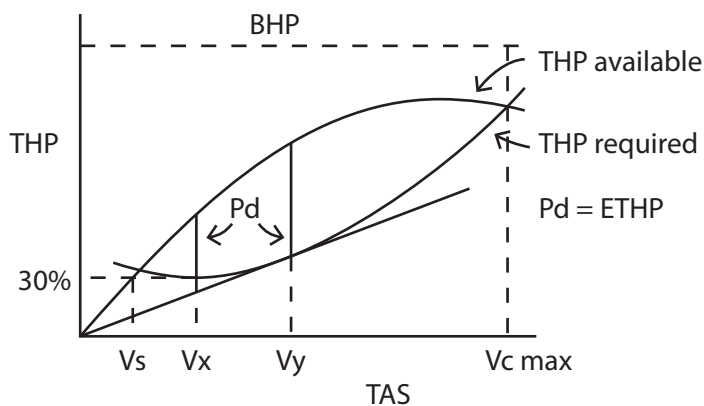
V = Knots true air speed

325 = constant.

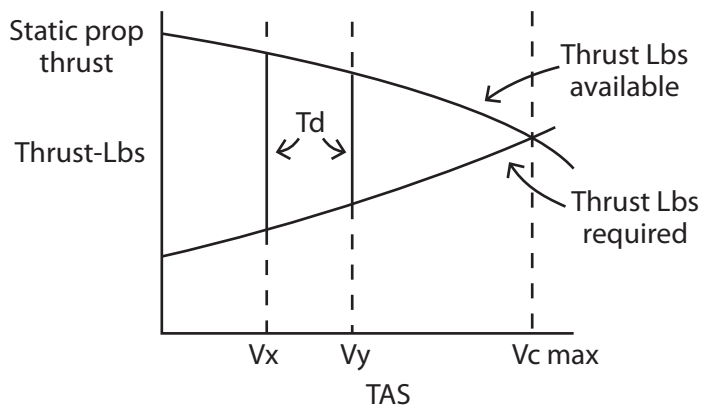
Note, the thrust horsepower available and thrust horsepower required curves converge at the stall speed, shown on **Diagram 60A**. The thrust-pounds required and thrust-pounds available curves show the greatest excess thrust occurs at zero speed and decreases with a speed increase, shown by **Diagram 60B**.

*Compare with Diagram 46, Jet-engine Power Curves. See Diagram 64, Power Curves – Helicopter.*

A)



B)



**Diagram 60, Thrust & THP Required & Available Curves**

**Pitch** Pitch is best described by a look at the definition of the term ‘pitch’ as applied to a propeller and its analogy with the common screw. When a screw is screwed into a solid medium such as wood, it will advance a given distance in each turn equal to its pitch; in other words, its advance per revolution is a fixed quantity equal to its pitch. The geometric pitch of the screw is the distance between each adjacent thread.

Now consider the case of a stationary aircraft with the engine running; the propeller’s advance per rev is zero. Next, consider an aircraft that is gliding with the engine and propeller stopped. The propeller’s advance per rev is infinite. Between these two extremes, the advance per rev of the propeller is a variable quantity depending on the speed, or lack of it, of the aircraft and propeller RPM. When the aircraft reaches a certain forward speed, the propeller blade’s angle of attack will become zero and the advance per rev will equal the geometric pitch, which is exactly what happens with a wood screw. The propeller’s geometric pitch angle of AB-AD is shown on **Diagram 38, Ground Effect Changes**. Three definitions of pitch can be applied to aerodynamics, thus:

- the aircraft’s pitch attitude is the up or down rotation of the aircraft’s attitude about the OY lateral axis.
- the rotor blade pitch is a helicopter’s main or tail-rotor blade angle relative to its axis of rotation. The blade pitch is measured at a defined station along the blade.
- The propeller pitch angle is the angle between the blade’s chord line and the plane of rotation, shown on **Diagram 68, Propeller Pitch** as angle AB-AD. It is also known as the blade pitch angle.

See **Diagram 37, Geometric Pitch**.

**Pitch angle** The angle between the chord line and the relative airflow of a propeller blade or rotor blade (AB-AD) as shown by **Diagram 37, Geometric Pitch**. It is also known as the blade pitch angle.

See **Blade pitch angle (BPA) & twist**.



**Pitch attitude** The pitch attitude is the angle between the reference plane and the OX longitudinal axis.

*See Pitch.*

**Pitch axis** The aircraft pitches nose-up or nose-down around the OY lateral pitch axis.

*See Lateral axis.*

**Pitch bucking** With the airplane in a stalled condition and the stick held fully back (up elevator) the nose may pitch up and down around the OY lateral axis. The rate of descent will vary as the aircraft stalls, recovers, and stalls again continuously until pilot in-put recovery is affected.

*See Stall break.*

**Pitch/diameter ratio** The ratio of propeller pitch to propeller diameter defines the pitch/diameter ratio.

The aircraft designer has to choose the propeller with the most suitable pitch and diameter for the aircraft and its intended mission and design air speed. When confronted with a family of propellers, which have their blade angles increased in some systematic order, one useful parameter to refer to is the pitch/diameter ratio. This ratio can be plotted on a graph, which shows the curve for the pitch/diameter ratio plotted against efficiency and the advance/diameter ratio, for a family of propellers with their pitch increasing. The propeller's efficiency increases with an increase of the ratio up to a certain limit. At too high a ratio, the angle of attack of the blades exceeds the critical angle at low forward speeds reducing the thrust available for take-off. Reducing the propeller's diameter also reduces the efficiency by over working the propeller blades.

**Pitch distribution** This is the changing blade angle (and hence the pitch) along the length of the propeller blade.

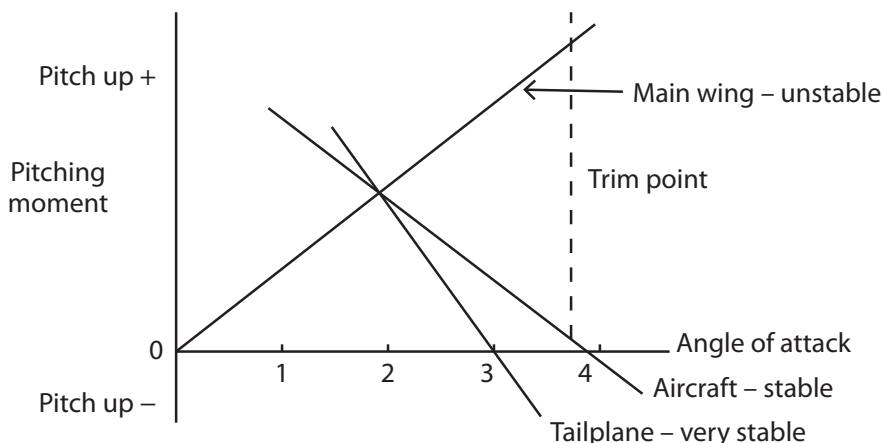
**Pitching** Activation of the elevators (or cyclic control in a helicopter) causes the aircraft's nose to rise and fall (pitch) around the OY lateral axis.

**Pitching moments** The conventional forces acting on an aircraft in straight and level flight are the lift/weight couple producing a nose-down pitching moment and the thrust/drag couple producing a nose-up pitching moment. The pitching moment is about the OY lateral axis and is positive when the nose pitches up. The tail-plane usually provides a tail-down load to balance any residual pitching moment, although in some instances, the tail-load force may be upwards.

For any given angle of attack and air speed, the wing pitching moments will be a constant value and to provide longitudinal stability the pitching moments must decrease as the angle of attack increases until the pitching moments are zero when the aircraft is in trim. With reference to *Diagram 61, Pitching Moments V. Angle of Attack*, consider the situation with the aircraft in cruise flight and in trim at  $4^\circ$  angle of attack with the pitching moments at zero. An unwanted pitch-up for whatever reason, say to  $6^\circ$  angle of attack will produce a nose-down pitching moment to return the aircraft to its trim speed. The same applies if the nose pitches down to say  $2^\circ$  below the trimmed angle of attack of  $4^\circ$ ; a positive nose-up pitching moment will be produced to restore the aircraft to its trim speed. In either situation, pitch up or down, the wing pitching moment will return the aircraft to its original angle of attack and air speed.

The main wing is inherently unstable, as shown by the up-slope line on *Diagram 61, Pitching Moments V. Angle of Attack*. The tailplane, due to its distance and surface area from the centre of gravity is extremely stable, shown by the steep down-slope on the graph. The tailplane moments counter the main wing moments to stabilize the aircraft, shown by the medium slope aircraft stable line on the graph.

An aircraft with positive stability will be shown with a negative slope of pitching moments (downward to the right) which decrease with increasing angle of attack. A steeper line shows greater stability.



**Diagram 61, Pitching Moments V. Angle of Attack**

**Pitching moment coefficient** The pitching moment coefficient ( $C_M$ ) is a dimensionless measure of the moments acting on an airfoil.

These moments are the pitching moments about the aerodynamic centre – the point on the airfoil's chord line at which the pitching moment coefficient for a symmetrical airfoil is zero because it does not vary with changes in angle of attack. However, for a cambered airfoil it is negative when the centre of pressure acts through the aerodynamic centre.

The pitching moment coefficient is relevant in the study of the aircraft's longitudinal stability. The pitching moment is equal to force times distance and is therefore a moment in itself. The formula is similar to the lift and drag formulas but it includes the

chord length. The dynamic pressure is a force equal to  $\frac{1}{2} \rho V^2$ , and wing area and wing chord length must be included in the equation for the pitching moment coefficient thus:

$$C_M = \frac{\text{Moment a.c.}}{\frac{1}{2} \rho V^2 S L}$$

Where  $\frac{1}{2} \rho V^2 = q$  = dynamic pressure

S = wing area

L = chord length

$C_M$  = moment coefficient

Moment a.c. = moment about the aerodynamic centre.

See **Aerodynamic centre (a. c.); Symmetrical airfoil; Cambered airfoil.**

**Pitch-up** A pitch-up is a positive movement about the aircraft's OY lateral axis upsetting the static longitudinal stability at increasing angles of attack.

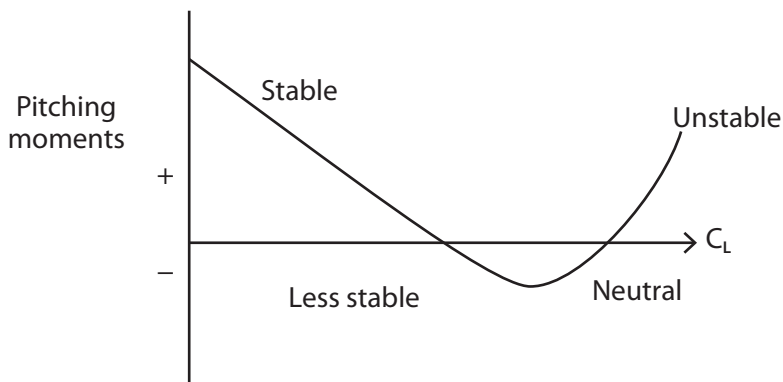
Positive longitudinal static stability is a requirement for safe, stable flight and is present at low lift coefficient/angles of attack, which is shown by **Diagram 62, Pitch-up Stability Curve**, by the downward, negative slope line. At higher angles of attack, the slope levels out showing neutral static stability before climbing upwards as a positive line, to show a negative stability situation. At this point, large pitching moments come into play developing high lift coefficients causing the nose to pitch-up, which in turn causes a further increase in lift coefficient and so on.

Pitch-up can be caused by several factors including:

- airflow separation and a stall at the wingtips of swept wing aircraft causes the centre of pressure to move forwards
- downwash from main wings or fuselage cross-flow on a horizontal T-tail produces unstable condition and decreasing stability

- flying at high angles of attack can induce pitch-up due to unstable centre of pressure movement, particularly on swept wing aircraft.

See **Fuselage cross flow**.



**Diagram 62, Pitch-up Stability Curve**

**Pitot inlet** The most basic jet engine inlet is a hole at the front of the jet engine known as pitot inlet, which works well up to about Mach 1.5. However, above that speed the inlet must be modified to allow for shockwave formations, which affect the airflow.

**Pitot-static system** The Pitot-static system consists of the Pitot tube and the static tube, which is used for measuring dynamic and static air pressure respectively, and is used on all aircraft for the operation of the air speed indicator, vertical speed indicator, and altimeter.

The open-ended Pitot tube faces into the free air stream. There is no internal velocity in the Pitot tube; the air stagnates in the tube causing total ram (or total head) pressure throughout the tube, which changes with changes in aircraft flight speed.

The static ports, which may be located concentric around the Pitot tube, or commonly on each side of the fuselage, measure

the free stream static air pressure. The Pitot tube and static ports are connected to a differential pressure gauge within the air speed indicator; this gauge measures the difference in dynamic pressure and static pressure to read free airstream dynamic pressure within the Pitot system, as indicated air speed. [Total pressure minus static pressure is equal to dynamic pressure]. The aircraft's altimeter and vertical speed indicators work off the static vent in the Pitot-static system, measuring changes recorded by the static pressure only.

The French hydraulic engineer Henri Pitot (1695–1771) invented the Pitot tube in 1732. He used his device to measure the River Seine's flow velocity.

See **Static air pressure; Bernoulli's theorem.**

**Plane of rotation** The plane of rotation is a reference line at right angles to the axis of rotation at which the total rotor thrust or propeller thrust acts. It represents the direction in which the propeller or rotor blades rotate. For a propeller, the plane of rotation and tip path plane are considered coincidental, because it is assumed the propeller does not cone upwards the way rotor blades do. On a helicopter's rotor disc, the tip path plane is considered to be above and parallel to the plane of rotation when under power due to the rotor blades coning upwards. The coning angle can therefore, be measured with reference to either the plane of rotation or the tip path plane.

The plane of rotation is shown on *Diagram 68, Propeller Pitch*, as the vector A-B.

Compare with **Tip-path plane.**

**Plane of symmetry** The plane of symmetry contains the aircraft's OX longitudinal axis and the OZ vertical axis. These two axes together determine the left and right side of the aircraft.

**Planform** When viewed from above, the planform refers to the geometric shape of the wing. The planform can be divided into low or high aspect ratio wings as listed below. Note, some planforms may be known by more than one name.

**High aspect ratio wings:**

Compound tapered  
Constant chord  
Crescent wing  
Flying wing  
Forward sweep  
Gull & inverted gull  
Oblique  
Reverse taper  
Slender  
Straight  
Sweptback  
Tapered  
Variable geometry  
Annular

**Low aspect ratio wings:**

Canard  
Delta  
Double delta  
Gothic delta  
Lifting body  
Ogival/Ogee delta  
Ogive delta  
Slender delta  
Trapezoidal

See **Aspect ratio**.



**Photo:** A Hawker Hunter suspended from the roof of the RAF Museum Cosford, UK, shows its swept wing planform.

**Planform coefficient** *See* **Lift coefficient**.

**Pods** Streamlined fairings on the trailing edge of the main wings known as pods, smooth the airflow over attachments such as flap track fairings, etc.

*See* **Anti-shock bodies; Flap track fairings; Küchemann carrots**.

**Point of reattachment** If the wing's boundary layer is tripped by a separation bubble, the flow may reattach to the wing's surface aft of the bubble. This point is known as the point of reattachment.

*See* **Flow separation; Separation point & flow**.

**Polhamus suction analogy** The Polhamus suction analogy is associated with vortex lift and was introduced in 1971 by E. C. Polhamus (a NASA Langley aerodynamicist). His analogy explains how the vortex lift on low aspect ratio highly sweptback wings at high angles of attack replaces the lift that is lost due to leading edge suction, which is present in a normal flow but is replaced by a vortex from the leading edge in a separated flow.

*See* **Vortex lift**.

**Positive acceleration** An aircraft is considered to have positive acceleration when it moves to the right along the OY lateral axis, upward along the OZ vertical axis and forwards along the OX longitudinal axis.

Positive 'g' is also a positive acceleration.

**Positive pitch** Pitching around the OY lateral axis is considered positive when the aircraft's nose pitches upwards.

A propeller blade's pitch is considered positive when it is set for normal forward flight, as opposed to braking pitch used for reverse thrust braking after touchdown.

*See* **Pitching moments; Propeller forces in cruise flight**.



**Positive pressure gradient** *See Adverse pressure gradient.*

**Positive roll moment** An aircraft rolling about the OX longitudinal axis is considered to be performing a positive roll when it rolls right-wing-down to the right.

**Positive static stability** An aircraft with positive static stability will seek its original trim speed following an upset due to turbulence or a control input by the pilot. The degree of positive static stability is dependent on the tailplane's restoring pitching moment. Therefore, the location, distance, area, and longitudinal dihedral of the tailplane are an important design feature of the aircraft to ensure positive static stability.

*See Stability; Static stability.*

**Positive stall** A positive stall is one entered with positive 'g' force (1g or more) from a non-inverted attitude.

*See Stall.*

**Positive yaw** An aircraft yawing about the OZ normal axis is considered to be performing a positive yaw when it yaws to the right.

**Potential & kinetic energy** Of the two types of energy, potential energy (the energy of position) is the energy that has the potential to do work, hence the name. In other words, it is the energy that has not yet been used (it is stored). For example, potential energy is always present in a descending mass, such as a waterfall, pile driver, etc. Kinetic energy (the energy of motion) is the energy possessed by a moving mass coming to rest or energy that has been or is being used, for example a hammer blow.

Note the similarities, or differences in the formulas below for kinetic energy and momentum:

$$\text{Kinetic energy} = \frac{1}{2} Mv^2$$

$$\text{Momentum} = Mv$$

The formula for the potential energy of a falling mass is:

$$\text{Potential energy} = Mgh$$

Where M = mass (kg)

g = acceleration due to gravity (9.8 m/sec<sup>2</sup>)

h = height in meters

V = velocity

When potential energy is converted to kinetic energy, the following can show:

$$(\text{PE}) Mgh = (\text{KE}) \frac{1}{2} Mv^2$$

Potential and kinetic energy has many applications in the physical world, however it is of interest to us in the aviation world, in relation to aircraft performance.

When the aircraft is stationary on the ground the energy is zero. When the engine power is increased for take-off, the aircraft accelerates and climbs and the kinetic and potential energy increases. Kinetic energy is stored in the aircraft's air speed and engine power. Potential energy is gained with an increase in altitude. At the aircraft's maximum straight and level speed, energy is equal to the lift and drag and the speed remains constant, although some energy is lost (transferred) in providing lift and the inevitable drag. In a descent, some kinetic energy from the engine and some potential energy from the decreasing altitude are again lost with potential energy being transferred to kinetic energy. Therefore, the total energy depends on air speed, engine power, and altitude. As we fly the plane through all normal manoeuvres, kinetic and potential energy is being used and transferred at all time without us being aware of the fact. This is the conservation of energy.

The important point here is to remember that energy can neither be destroyed nor created but energy is converted to heat, sound

and disturbance from air (drag), etc. Energy is a scalar quantity not a vector; direction is not considered but speed is.

Potential energy is measured in foot-pounds (ft-lbs).

*See* **Kinetic energy; Bernoulli's theorem; Conservation of Energy.**

**Potential flow** A potential flow describes the velocity field of an incompressible flow around an airfoil at low sub-sonic speeds, below 200 KTAS/Mach 0.3.

It is a theoretical description of the flow around an airfoil, but it ignores the effects of the boundary layer and air viscosity. The potential flow is characterized by an irrotational (zero vorticity) flow, and is valid for several applications, including the outer flow fields around airfoils.

The potential flow satisfies Laplace's equation, which was introduced by Pierre Simone Laplace (1749–1827) the French mathematician and theoretical physicist.

**Pounds of pull** The pounds of pull are also known as 'stick force per 'g' per knot lost'.

With an increasing load factor (g) during a steep turn, the pounds of pull increases. This means as the yoke is pulled rearwards to raise the nose (or to maintain height in a turn) with decreasing air speed the elevator force gets heavier.

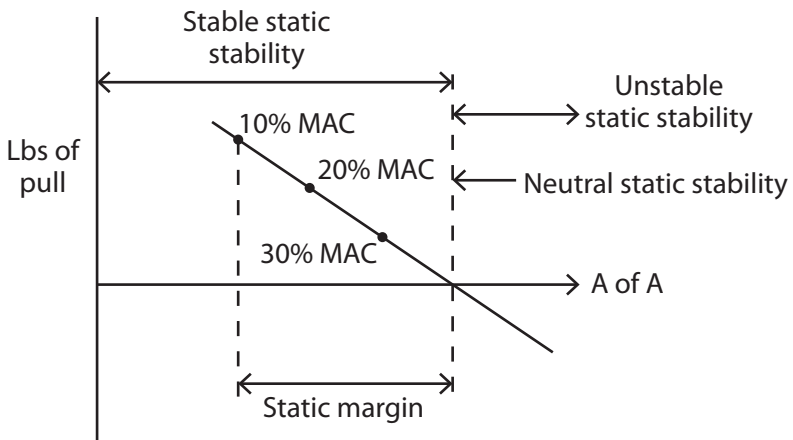
When the centre of gravity is located at the neutral point (the point of neutral static stability), the required pounds of pull to raise the nose is zero. As the centre of gravity location is moved further forward from the neutral point to 30% MAC and then to 20% MAC, the pounds of pull required raise the nose will gradually increase. At the forward centre of gravity limit, the nose will be harder to raise due to the increased stick force required and the increased static stability. The pounds of pull required to move the stick should increase linearly with forward centre of gravity movement resulting in a straight line on the graph. However, it is

acceptable for the line to vary slightly from being straight for some aircraft.

With an increasing load factor ( $g$ ) during a steep turn, the pounds of pull increases. Manoeuvring stick force gradient or stick force per ' $g$ ' must be positive for static stability but in addition, a high a force makes the aircraft hard to handle. Fighter aircraft have light stick forces of about 3–8 pounds of pull for high combat manoeuvrability. A jet transport has heavier stick force due to its greater stability and because of its limit load factor being a great deal less than a fighter aircraft. Static stability, stick force and manoeuvring stability are all related.

Altitude also affects the pounds of pull required for manoeuvring. At high altitude where the air density is lower, the pounds of pull is less than that at sea level, due to the higher true air speed producing less downwash on the tailplane, which reduces the longitudinal pitch damping moment.

Reference to **Diagram 63, Pounds of Pull V. CG Percentage MAC**, shows the relationship between the force required on the control column to raise the nose (pounds of pull) and the centre



**Diagram 63, Pounds of Pull V. CG Percentage MAC**

of gravity (CG) expressed as % MAC. The diagram also shows the neutral static stability line, which is the dividing line between the positive (stable) and negative (unstable) static stability. The static margin is measured from the neutral static stability line (a fixed point) to the centre of gravity location (a variable point).

See **Stick force per 'g'**.

**Power** Power is the rate of doing work or, the rate of transfer of energy; it is measured in ft.lbs per second shown by the formulas as follows:

$$\text{Power} = \frac{\text{work}}{\text{time}} = \frac{\text{force} \times \text{distance}}{\text{time}}$$

The amount of work an aircraft engine can do in a given amount of time is dependent on how much, and how quickly the chemical energy from the fuel can be transformed into mechanical energy. The English system uses the term brake horsepower, which is the more common term in the aviation industry, with the metric term of Watts being used less frequently, where:

$$\begin{aligned} 1 \text{ HP} &= 33000 \text{ foot-pound per min} \\ 1 \text{ Watt} &= 1 \text{ Joule/sec} \\ 1000 \text{ Watts} &= 1 \text{ Kilowatt.} \end{aligned}$$

**Power absorption** The amount of engine brake horsepower a propeller can absorb and convert to thrust depends on a variety of factors – the propeller's diameter, number of blades and the blade's aspect ratio.

**Power curves – helicopter** The helicopter thrust horsepower required curves are a similar shape to the drag curve, shown on *Diagram 20, Total Drag Curves*. However, note the power curves are plotted against true air speed and the drag curves are plotted against indicated air speed. The curves for parasite, induced and

profile power are drawn below the total thrust horsepower required curve.

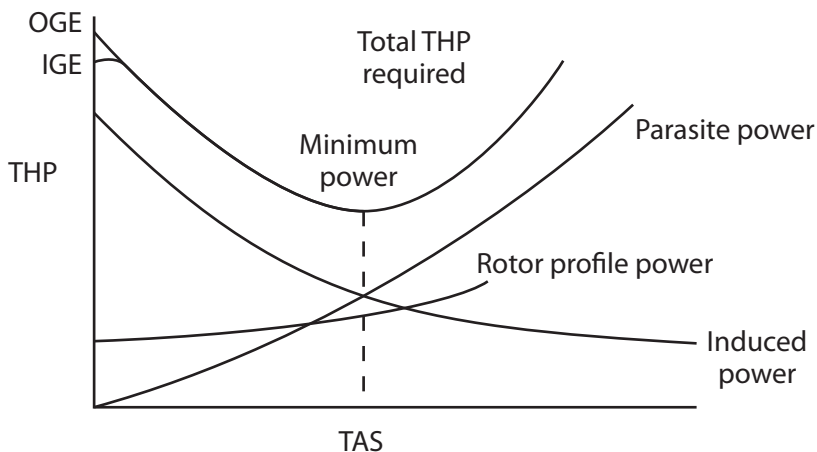
The induced power is at a maximum during the hover at zero forward speed. It decreases with an increase in forward speed due to the reduction in the inflow angle and main rotor drag. Induced power is the power required to drive the main rotor blades to provide total rotor thrust (lift) to equal helicopter weight.

Translational lift accounts for the decrease in induced power required. It occurs when the helicopter moves forward from the hover into accelerating flight. The induced inflow changes from a vertical flow during the hover to a more prominently horizontal flow as the forward speed increases through 10–15 Knots up to cruise speed. With increasing inflow velocity due to the increasing forward speed, the total rotor thrust increases and less power and collective (reduced angle of attack) is required than in the hover.

Profile power is that portion of total power used to maintain the main rotor RPM and tail rotor power drive plus the power required to drive the ancillary equipment.

A helicopter moving forwards at low-speed has approximately the same airflow-speed over both the advancing and retreating rotor blades. However, due to the speed-squared factor ( $V^2$ ) becoming more apparent with increasing forward speed, the difference in drag between the advancing and retreating blades is appreciable, even though the rotor RPM remains constant. Therefore, the profile power required to overcome the profile drag rise increases with helicopter forward speed.

The parasite drag produced by the helicopter's fuselage increases in proportion to the forward velocity squared ( $V^2$ ). It requires the main rotor disc to tilt forward to increase the forward thrust component of the total rotor thrust in order to overcome the increasing parasite drag. With the disc tilted forwards, the inflow angle increases and the total reaction tilts rearwards, which increases the rotor drag. The angle of attack reduces, which reduces the lift vector and so an increase in total rotor thrust is required.



**Diagram 64, Power Curves – Helicopter**

The total thrust horsepower required curve is the sum total of all power required. The curve has two commencing lines at the left-hand edge of the graph; the lower line (IGE) shows that less total thrust horsepower is required when hovering 'in ground effect' as opposed to the upper line (OGE) which extra power is required when hovering 'out of ground effect'.

The total power required curve then dips down with increasing forward speed (due mainly to induced drag decreasing before curving upwards due to parasite drag increasing when more thrust horsepower is required with increasing air speed. Minimum power occurs where the parasite and induced power curves coincide and the same value. All speeds are true air speed.

See **Translational lift; Profile drag.**

**Power differential** The difference between thrust horsepower available and thrust horsepower required is the power differential.

See *Diagram 60, Thrust & THP Required & Available Curves.*

**Power loading** The power loading (weight-to-power ratio) is the ratio of aircraft weight – usually given as the maximum take-off weight – to engine brake horsepower. The value is approximately 10 to 20 pounds per horsepower for average light aircraft. Aircraft performance improves with lower power loading; that means shorter take-off distance and better rate of climb. Thrust loading is the term used for gas turbine (jet) engines.

**Power-on stall** A power-on stall, either intentionally or by accident is one with engine power applied. During an intentional stall for practice, the nose is gradually raised as speed decays and at an increased angle of attack, the slipstream over the wings helps provide some lift, resulting in a few Knots reduction of stall speed. The stall break is more pronounced than a power-off stall followed by a rapid nose-down pitch movement.

*See Stall.*

**Powered slat** Some light aircraft (STOL equipped) have slats, which extend due to the suction on the leading edge of the wing when the angle of attack increases to a pre-set value. However, on larger aircraft, jet transports for example, the leading edge slats are extended under power with initial trailing edge flap application.

*See Leading edge devices (LEX); Slats.*

**Power settling** Power settling is an alternative name for settling with power, a condition associated with vortex ring state. A helicopter descending vertically can be caught in its own rotor downwash, which further increases its rate of descent to the point where a heavy landing results.

*See Settling with power; Recirculation; Vortex ring state.*



**Prandtl, Professor L.** The greatest aerodynamicist of all time must be Professor Ludwig Prandtl (1875–1953) who was born in Freising, Bavaria. His studies commenced at the Technische Hochschule (university) in Munich, Germany in 1894, graduating in 1900 from the University of Munich. His early studies involved solid mechanics. In 1901, he became a professor at Hanover with a worldwide reputation in the study of fluid mechanics, the basis of aerodynamics and hydrodynamics.

In 1932, Professor Prandtl worked alongside Albert Betz (1885–1968) and Maxwell Munk (1890–1986) to solve the problems of wingtip vortices and how they cause induced drag and their influence on the wing's pressure distribution. The conclusion of these studies was his lifting line theory, which he published in 1918–19. The lifting line theory developed by Professor Prandtl, for calculating the initial performance of finite wings is still used by aerodynamicists today. In recognition of Dr. Frederick Lanchester's (1868–1946) earlier work on the lifting line theory, the principle is now known as the Lanchester-Prandtl lifting line theory.

In 1904, he presented his boundary layer theory and the mathematics to prove his point. His theory studied how the airflow changed across the wing from streamlined (laminar flow) to turbulent boundary layer with its separation causing drag and explained the stall, now common knowledge to all pilots. He also worked alongside his colleague Professor Herman Schlichting (who also takes much credit for the study of the boundary layer phenomenon) on the effect of skin friction and the associated Reynolds number. In Heidelberg during 1904, Professor Prandtl presented to the Third Congress on Mathematics his ideas on boundary layer flow. He moved in 1904, to the University of Gottingen where he became the Director of Institute of Technical Physics. His laboratory was noted worldwide for being the greatest aerodynamics research facility between 1904 and the 1930s.

His initial interests in supersonic flow commenced during his early years of research (1906–7) with one of his students, Theodore Meyer. Together, they developed the theory for calculating

shockwave properties in supersonic flow in 1908, with the use of schlieren photography.

During the 1910–20 period, Professor Prandtl established the science of modern aerodynamics, as pilots know it today, plus many other aerodynamic theories. For example, from 1911–1918, he developed airfoil and wing theories, the lifting line theory for finite wings and how the centre of pressure moves forward with an increase in angle of attack on cambered wings and the fact it always lies behind the 25% chord location at normal angles of attack. [Samuel Langley (1834–1906) noted it moved the opposite way on a flat plate]. Prandtl returned to the supersonic flight theory in 1929 when he solved the solution with Dr. Adolph Büsemann (1901–1986) on designing supersonic nozzles. He also introduced the biplane interference factor, used for calculating the loss in lift due to the positive pressure on the lower wing interfering with the negative pressure of the top wing on a biplane.

Professor Prandtl's many and varied aerodynamic projects advanced the science of aerodynamics to what we know and accept today. In the 1930s, he was bestowed the titles of 'elder statesman' and 'father of aerodynamics' and 'world leader in aerodynamics'. Although he never received the Nobel Prize for all his contributions to the science, he should have – he deserved it!

One of Prandtl's students was Dr. Adolph Büsemann (1901–1986) who introduced the idea of using sweptback wings for high-speed aircraft. Professor Prandtl presented his theory to the world on 30 September 1939 at the Volta conference in Rome, Italy on the effect of compressibility on high-speed flight. It is claimed, he built the first supersonic wind tunnel specifically for his research on supersonic flight in the mid-1930s. However, it is also claimed he was using a small supersonic wind tunnel as early as 1905 for his shockwave research.

**Prandtl-Glauert compressibility correction**  
**Compressibility correction.**

*See*

**Prandtl-Glauert condensation cloud** The airflow over an aircraft is always subject to changes in air pressure, particularly over the wings. A drop in temperature is associated with a sudden drop in pressure. When the air is sufficiently humid, the temperature drop will cause the moist air to form condensation clouds under two different flight conditions. The first condition is evident at relatively low air speeds such as during take-off or landing and during high-g manoeuvres as lift-induced condensation. The condensation cloud is asymmetric and roughly an elliptical shape forming mainly on the low-pressure, upper-side of the wing. Photographs of Concorde on take-off or landing show this clearly.

The second condition for formation of a condensation cloud is found on jet fighter aircraft flying in the transonic region. In this flight condition, a symmetrical condensation cloud is formed due to the sudden drop in air pressure and temperature through the shockwave formed by the aircraft. Due to the compressibility present an N-wave is formed over the aircraft. The condensation cloud then has the appearance of a round (front view) and diamond shape (side view) small cumulus cumulus type cloud, with the aircraft flying through it, although the cloud is attached to and moves with the aircraft.

This phenomenon is known as the Prandtl-Glauert condensation cloud (or singularity) after the two aerodynamicists (Prandtl and Glauert) who first explained it.

They are also known as shock collars, shock eggs and N-wave.

See **Prandtl-Glauert singularity**.

**Prandtl-Glauert equation** The Prandtl-Glauert equation is used to correct the coefficients of lift, drag, and pressure at high-subsonic speeds to be equivalent to the same values at a lower speed where the effects of compressibility can be ignored.

**Prandtl-Glauert interference factor** The Prandtl-Glauert interference factor is used to calculate biplane interference – the interference of lift and the calculation of the induced drag caused by the close proximity of the upper and lower wings of a biplane. The interference factor assumes the wings are elliptically loaded and is used to calculate the wing downwash interference factor caused by the vertical gap between the two sets of wings plus their span ratio.

$$\frac{\text{Gap}}{\text{Mean span}} \quad \text{and} \quad \frac{\text{upper span}}{\text{lower span}}$$

It is also known as the Biplane Equation.

**Prandtl-Glauert law** The Prandtl-Glauert law allows for the effects of compressibility on a two-dimensional airfoil section in subsonic flow. Professor Ludwig Prandtl (1880–1953) in 1922 and Herman Glauert (1892–1934) in 1927 were both studying the theoretical effects of compressibility.

*See Prandtl, Professor L.; Glauert, H..*

**Prandtl-Glauert singularity** The formation of the Prandtl-Glauert condensation clouds is explained as a mathematical singularity of aerodynamics, known as the Prandtl-Glauert singularity.

*See Prandtl-Glauert condensation cloud.*

**Prandtl-Meyer expansion fan** When a supersonic flow rounds an expansive (convex) corner it produces an expansion fan and flow where the temperature, density, and pressure all decrease. The corner may be curved or sharply angled and a supersonic flow will follow it through an expansion fan, unlike a subsonic flow, which would breakaway and cause turbulence. Prandtl and Meyer discovered the phenomena in 1907–8.

*See Shock cloud; Expansion flow & fan; Mach lines.*

**Precision spin** A precision spin is one performed intentionally and as accurately as possible as a flight exercise manoeuvre. The spin is entered with positive control inputs and must be stabilized within a quarter turn. Recovery must be achieved within a certain number of turns.

*See Inverted spin; Normal spin; Spinning,*

**Pressure** In physics and meteorology, atmospheric pressure is defined as the amount of force per unit area (or weight) of a vertical column of air acting perpendicular to the Earth's surface. Pressure is a scalar quantity – it has no direction. It is the force produced by the random motion of the molecules in the air mass acting in all directions, which produces static pressure as read by an aircraft altimeter or barometer. The term static pressure should be used when referring to pressure to differentiate from dynamic or total pressure, which is used in Bernoulli's formula.

The atmosphere is held in place around the earth due to gravitational attraction; the action of the molecules in the air prevents the atmosphere from collapsing. The atmosphere therefore has its maximum pressure at sea level due to the weight of the air above it. In 1644, Evangelista Torricelli (1608–1647) an Italian mathematician and physicist who made the first barometer, proved that the atmosphere surrounding the Earth exerts a pressure due to the weight of air, which decreases with altitude. Torricelli's first experiments involved a tube, closed at the top, open at the bottom end and placed in a container of water. The air pressure supported the water in the tube, which stabilized at around 10.3 meters (34 feet) in height. This same experiment was repeated using mercury in place of water and because mercury is 13.6 times denser than water, the mercury stabilized at approximately 0.76 meters (2.5 feet) or one atmosphere. This was the first barometer of the mercury type.

In the 1840s, the French invented the aneroid (capsule) barometer. [Aneroid is taken from the Greek word of 'a neros'

meaning without fluid]. In addition, Baros means weight and metros means measure. The aneroid barometer is now the most common type in use and it forms the basis for the altimeter, air speed indicator and the vertical speed indicator we used as flight instruments, which measure the air pressure via the Pitot-static system. The altimeter measures the static or ambient air pressure and the vertical speed indicator measures the rate of change of the static air pressure during a climb or descent. The air speed indicator measures the difference between static and dynamic pressure.

The SI unit for standard pressure is the Pascal (Pa) after Blaise Pascal (1623–1662):

- Newton per square meter ( $\text{N/m}^2$ )
- Pascal ( $1 \text{ N/m}^2$ )

The Imperial units of pressure are:

- $\text{Lb./ft}^2$
- $\text{Lb./in}^2$  (PSI)
- $1 \text{ Atmosphere} = 760 \text{ mmHg} = 14.7 \text{ lb./in}^2 \text{ (PSI)} = 29.92 \text{ inHg} = 1013.25 \text{ hPa}$ .

Hectopascals (hPa) previously known as millibars, are used on synoptic weather charts and for altimeter settings.

*See **Absolute pressure.***

**Pressure altitude** The pressure altitude is that indicated by an aircraft altimeter set to a barometric pressure different to the standard atmosphere of 29.92 inches Hg or 1013.25 hPa (or mbar) and is known as the QNH setting. The altimeter then indicates the vertical height (pressure altitude) at any level in the International Standard Atmosphere. Variations in temperature, pressure, and lapse rate in the atmosphere may cause the altimeter to not read the actual height above sea level but instead, an indicated altitude, which corresponds to a particular pressure altitude.

Pressure altitude is akin to density altitude, being a high, or low-pressure altitude. A high pressure (or density) altitude are adverse conditions for aircraft performance; a low pressure (or density) altitude produces favourable operating conditions.

*Compare with Density altitude.*

**Pressure coefficient** The pressure coefficient ( $C_p$ ) is the ratio of the relative change in static air pressure compared to the dynamic air pressure measured on the surface of a wing. It represents the change in air pressure above and below the wing and thus, the lift and drag forces. Ambient static pressure records a pressure coefficient of zero as opposed to a pressure coefficient of one (1) for dynamic pressure (as measured by a Pitot tube) in an incompressible low-speed flow. In a high-speed, compressible flow, the pressure coefficient is greater than one.

The value of the pressure coefficient on the wing's upper surface determines the maximum flow speed, the increase of pressure in the adverse pressure gradient and an indication of incipient shock wave formation.

Pressure coefficients versus a chosen angle of attack can be plotted on a graph. The plots above the zero base line indicate the reduced air pressure on the wing's upper surface. Conversely, the plots below the line represent positive pressure on the wing's lower surface.

The pressure coefficient is given by the following formula:

Static (relative) pressure ( $p - p_o$ ) divided by the free stream dynamic pressure ( $q$ ).

$$C_p = \frac{(p - p_o)}{q} = \frac{\text{static pressure}}{\frac{1}{2} \rho V^2}$$

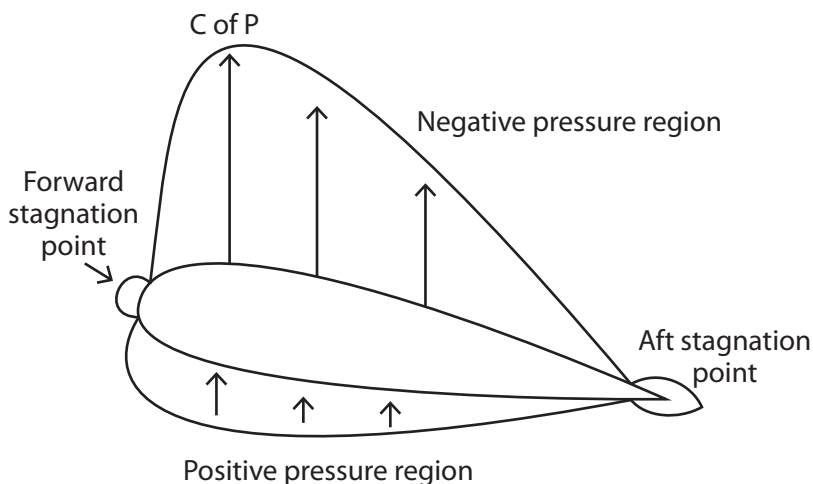
Where  $C_p$  = pressure coefficient

$(p - p_o)$  = static pressure

$q$  = dynamic pressure  $\frac{1}{2} \rho V^2$ .

Note, the static pressure symbol is 'p' and the dynamic pressure symbol is 'q'.

## Pressure cowling *See NACA cowling.*



**Diagram 65, Pressure Differential & Stagnation Points**

**Pressure differential** The pressure differential between the upper and lower surfaces of a wing produces an aerodynamic reaction known as lift. Reference to **Diagram 65, Pressure Differential & Stagnation Points**, it can be seen the air pressure over the upper surface is reduced to that below the ambient air pressure and increased above ambient on the lower surface (most times). The difference in pressure is due to the shape of the airfoil section, angle of attack and the induced upwash in front of the airfoil and the downwash behind it. The maximum pressure differential occurs at the wing's point of greatest thickness (or maximum camber). The pressure over and under the wing is known as gauge pressure, when compared to the free-stream airflow pressure.

The average, sea level ambient air pressure is about 14.7 pounds per square inch with the air pressure above the wing being slightly less (about 14.4 pounds per square inch) and below the wing, it is slightly greater (about 14.9 pounds per square inch). A propeller's pressure differential is the difference in pressure between the



front and rear of the propeller disc; it is this pressure differential that produces thrust.

The free stream static pressure is represented by the symbol ( $p_o$ ) and the local pressure on the wing's surface is represented by ( $p$ ). Therefore, the pressure differential is shown by ( $p_o - p$ ).

**Pressure distribution** The pressure distribution over and under a wing can be represented by a pressure-plotting diagram, which shows the changes and extent of the plot with changing angle of attack. The diagram will show a decrease of pressure above the wing and a corresponding increase below the wing. The pressure is unevenly distributed with most of the pressure decrease and pressure increase (above and below the wing respectively) occurring forward of the quarter chord point on the average light aircraft wing. This shows the forward quarter section of the wing produces about half of the lift force. At the leading edge of the wing, the airflow is brought to a standstill at the stagnation point and then the airflow velocity builds up rapidly over and under the wing. This is also known as airfoil loading.

*See* **Lift distribution; Stagnation point/line.**

**Pressure drag** Consider a flat plate normal to a flowing stream of air. The pressure in front of the plate will be greater (over pressure) than the pressure behind it (under pressure). The drag of a flat plate can be considered as being a maximum at 100% with a reduction in drag as the flat plate is streamlined.

Pressure drag acts on the wing's leading edge and just below it; the decreased pressure counteracts this over the rear portion of the wing. Pressure drag is also present in the wing's wake due to the increased turbulence downstream of the separation point.

Pressure drag (also known as form drag) is also a part of the wave drag caused by pressure forces in the boundary layer in a supersonic airflow. A thickening of the boundary layer causes separation, which in turn, causes an increase in drag.

*See* **Form drag; Wave drag; Drag divergence curve.**

**Pressure envelope** The pressure envelope is a line on a diagram representing the boundary of a positive or negative area of pressure, as found for example, above or below an airfoil of an aircraft.

*See Diagram 65, Pressure Differential & Stagnation Points.*

**Pressure face** The term pressure face refers to the rear of the propeller blade or lower surface of a helicopter rotor blade where the pressure is greater than atmospheric. The relatively flat surface of the propeller blade corresponds to the wing's lower surface. On a single-engine aircraft, the pilot would see the pressure face when sitting in the aircraft with the prop/engine in front of him.

Also known as the thrust face.

**Pressure gradient** The pressure gradient is the difference in pressure of the thin layer of air flowing chordwise over the upper surface of the wing. The airflow initially increases in velocity with a decrease in pressure and then changes to a flow with decreasing velocity and rising pressure. Within the pressure gradient, the flow is initially laminar up to the transition point where it then changes to become a turbulent flow.

**Pressure law** *See Charles' and Gay-Lussac's Law.*

**Pressure recovery** *See Ram effect/recovery.*

**Pressure signature** An aircraft flying above Mach 1.0 (supersonic) forms two conical shockwaves, one each from the nose and tail, which may reach the ground. The distance between the two shockwaves is known as the pressure signature, which reaches the ground as an audible sonic boom. The longer the aircraft, the longer will be the pressure signature; Concorde's pressure signature is a double boom, a quarter second apart. The volume of the boom depends on the aircraft's altitude, weight, and atmospheric conditions within which it is flying.

*See Sonic boom; N-wave.*

**Pressure – static & dynamic** In aviation we frequently refer to fuel, oil, hydraulic pressure and the pressure of air. Nevertheless, what is said about air pressure here can also be applied to fuel, oil, and hydraulic pressure. Pressure can be further defined as being static or dynamic.

The physicist would describe pressure by using the molecule theory; every fluid, including air is made up of an infinite number of molecules. When the air is at rest, the molecules exert a uniform pressure in all directions due to their random motion. This uniform molecule pressure is known as static pressure.

Our bodies feel the static pressure of air. Some may argue, they cannot feel it but, air pressure at sea level is acting on our bodies (and everything else) with an average force of approximately  $101.3 \text{ kN/cm}^2$  (14.7 lbs./sq. inch). Our body produces an equal and opposite reaction or outward force; that is why we do not feel the air pressure.

If we are outside on a windy day, we can feel the force of the wind pushing on our bodies. The molecules of air are now traveling with the wind and we feel this as dynamic pressure. The same applies to the aircraft. When the aircraft is at rest on the ground and assuming no wind, only the static air pressure is acting on the aircraft. However, when the aircraft is in flight, the additional force of dynamic air pressure acts on all areas of the aircraft, in particular the 'nose' and wing leading edges. The dynamic air pressure has a force in Imperial units of 25 PSI at 100 MPH or 100 PSI at 200 MPH varying with the square of the indicated air speed. [Imperial units are used here as an example because the numbers are easier to follow]. Dynamic pressure of course, contributes to the aerodynamic reaction of the airfoils that is lift and drag.

**Pressure surface** The area under the wing where the pressure is relatively higher than over the wing is known as the pressure surface.

**Pre-stall buffet** Prior to reaching the critical angle of attack, the turbulent flow leaving the trailing edge of the main wing breaks away and flows over the tailplane. The pre-stall buffet can be felt as a shaking or buffeting throughout the airframe and the pilot's yoke or stick.

*See Burble/burbling.*

**Primary flight controls** The primary flight controls consist of the spoilers, ailerons, elevators and the rudder. High-lift and drag producing devices and trimmers are secondary.

*See Control surfaces.*



**Photo:** The Boeing 727 in flight showing its sweptback port wing with inboard and outboard ailerons and the four spoiler panels ahead of the slotted Fowler flaps.

**Primary structure** The aircraft's primary structure is that which would be a hazard to flight if that structure failed, e.g., wings, undercarriage, etc.

**Principal inertia axes** An aircraft's three principal inertia axes are the OX longitudinal, OY lateral and the OZ vertical axes. They are also known as the body axes, principle axes of inertia or aerodynamic axes.

*See Aerodynamic axes.*

**Principal of moments** A moment is the turning effect produced by a force about its axis.

The moment of a force is proportional to its magnitude or weight and the perpendicular distance from its axis to the line of action of the force. That is to say, a large weight located a short distance from the fulcrum can be balanced by small weight at a greater distance. For example, one weight of 10 kg placed 1 meter from the pivot point produces an anti-clockwise moment. A second weight of 5 kg placed at a distance of 2 meters from the pivot point that produces a clockwise moment thus:

$$10 \text{ kg} \times 1 \text{ m} = 10 \text{ kg-m, anti-clockwise}$$

$$5 \text{ kg} \times 2 \text{ m} = 10 \text{ kg-m, clockwise.}$$

The sum of the clockwise moments equals the sum of the anti-clockwise moments and maintains the system in equilibrium.

The principal of moments is of great importance when calculating the centre of gravity of an aircraft. The moments are found by multiplying the given weight of each item in kg or pounds (empty aircraft, fuel, baggage, and passengers by their respective moment arms in mm or inches thus:

$$\text{Weight} \times \text{arm} = \text{moment.}$$

The answer is given in kg-mm or in pound-inches. All items located forwards of the aircraft's datum become anti-clockwise or negative moments and these are subtracted from the clockwise moments that are located aft of the datum. The total moments divided by the aircraft's weight determines the centre of gravity location aft of the datum in centimeters or inches. By convention the aircraft's nose points to the left of the diagram.

*See Moment.*

**Profile drag** Pressure (or Form) drag and skin friction combined produce profile drag. Streamlining and a smooth surface help reduce profile drag. The wing's sharp trailing edge also helps to reduce profile drag by allowing the two airflows, over and under the wing to depart the wing at a similar angle to each other, ensuring a turbulent free flow as much as possible. The use of flat head rivets in place of round head rivets or surface bonding also contributes to reduced profile drag.

On a helicopter's rotor blade at low forward speed the increased profile drag on the advancing blade is counteracted by the decreasing profile drag on the retreating blade. Therefore, the drag almost cancels out – there is a slight rise with increasing forward speed. However, at high forward speeds, the speed-squared factor ( $V^2$ ) as found in the lift formula becomes more prominent and profile drag increases over the rotor blade, particularly towards the rotor blade tips.

Profile drag is also known as zero lift drag. It is related to 2-D research aerodynamics; when related to 3-D aerodynamics it is known as parasite drag.

*See* **Total drag; Boundary layer drag; Parasite drag; Profile power.**

**Profile drag power loss** *See* **Profile power.**

**Profile power** Because drag is equal to power, the power curves are of a similar shape to the drag curves. A helicopter moving from the hover to low forward speeds has approximately the same airflow speed and profile drag on the advancing and retreating blades. However, as the helicopter's speed increases, the speed-squared factor ( $V^2$ ) becomes more apparent and the difference in rotor drag is appreciable.

Profile power is that portion of total power required to oppose the effect of profile drag caused by the main and tail rotor systems. The advancing blade drag increases considerably compared to

the retreating blade drag. Therefore, the profile power required to overcome the profile drag rise increases with helicopter forward speed.

It is also known as profile drag power loss.

See **Profile drag; Power curves – helicopter.**

**Profile thickness** Profile thickness is an alternate name for the wing's depth, or thickness. It is the maximum depth of the wing between the upper and lower surfaces.

See **Thickness/chord ratio; Diagram 4, Airfoil Terminology.**

**Progressive stall** A progressive wing stall that occurs relatively gradually is a desirable quality. This is achieved by careful design where pre-stall buffet and a gentle stall break accompany the gradual breakdown of airflow over the wing.

*Diagram 52, Lift Coefficient V. Straight, Swept and Delta Wings*, shows the peak of the lift coefficient curve has a sharper peak (it is peaky) for a straight wing and is more rounded (a rooftop) for a sweptback wing. The sweptback wing has a more desirable progressive stall than a straight wing.

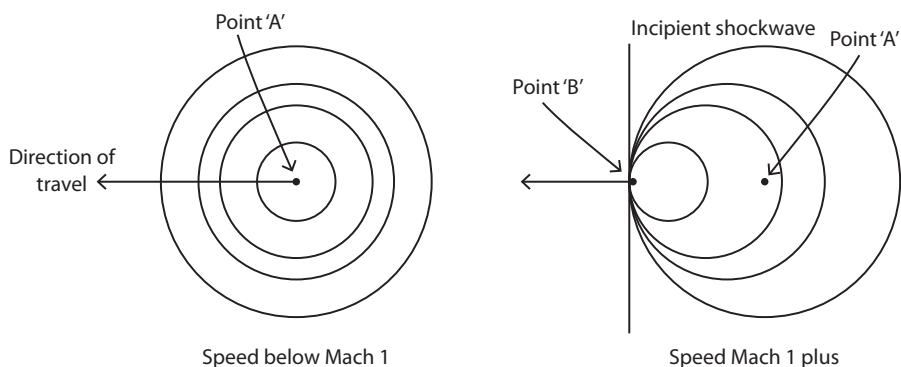
See **Stall.**

**Propagation of shockwaves** An aircraft in flight causes a disturbance to the air around it. This disturbance is transmitted in all directions at the speed of sound as pressure waves (or Mach waves).

*Diagram 66, Propagation of Shockwaves*, shows these pressure waves below Mach 1.0 as circles representing the movement of an aircraft (point 'A'). As the aircraft's speed increases to the critical Mach number, the pressure waves have less time to move ahead of the aircraft (they bank up like a highway traffic jam) until eventually the airflow over the wing reaches Mach 1.0, the same speed at which the pressure waves are moving (point 'B'). At

the critical Mach number, an incipient shockwave will form on the surface of the aircraft's wings, progressing into an advanced shockwave as speed increases further. The pressure waves merge into the Mach cone at high Mach numbers.

See **Mach angle & cone; Shockwaves.**



**Diagram 66, Propagation of Shockwaves**

**Propeller aerodynamics** Different authors have put forward several theories on propeller aerodynamics. The theories include the vortex, general momentum, axial momentum and the blade element theory. The blade element theory is the most widely accepted theory, along with parts of the axial momentum theory, leaving the vortex and general momentum theories to fall by the wayside.

In 1865, the Scottish engineer and scientist William George Rankine (1820–1872) founded the axial momentum theory while working on the theory of ship's propellers. At a later date further work and development on the axial momentum theory, was covered by Robert Edmund Froude (c1830–?) also an engineer. The blade element theory was first introduced by William Froude (1810–1879) the father of R. E. Froude, an engineer and naval architect in 1878, when he also was working on ship's propeller theories. Note the theory of ship and aircraft propellers is virtually



the same, because air at subsonic speed behaves very similar to flowing water.

Stefan Drzewiecke further developed the blade element theory from 1892 onwards and he has been credited with the majority of the research work. The axial momentum theory, also known as the Rankine-Froude theory after its two author's deals with the energy change given to the air mass after it passes through the propeller disc. It also includes the effect of the rotational slipstream, the friction drag of the propeller blades and the loss of energy in the slipstream caused by the interference of the engine nacelle or the fuselage, amongst other factors. The blade element theory deals with the forces acting on the propeller as it moves through the air at a uniform velocity. It also includes the blade's shape and number of blades and assumes the propeller blade to be made from an infinite number of blade elements, hence the name blade element theory. Theodore Theodorsen (1897–1978) of NACA also performed propeller research. The diagrams in this book represent the blade element theory, for example *Diagram 67, Propeller Forces in Cruise Flight*.

**Propeller back** This is the curved upper surface of the propeller and the side that is viewed from in front of the plane.

**Propeller blade drag** The propeller acts like any airfoil moving through the air, produces an aerodynamic reaction due to its profile shape, angle of attack and velocity. The total reaction can be divided into the vector components of lift and drag. It is the drag component we are interested in here.

Examination of *Diagram 67, Propeller Forces in Cruise Flight*, shows when the propeller is operating at its maximum lift/drag ratio it is producing the most lift for the least amount of drag. It follows the most thrust for the least amount of engine power used (the maximum thrust/torque ratio) will also be achieved at the maximum lift/drag ratio condition, and it will be operating at its maximum efficiency.

When the propeller blade is advancing through the air on its helical flight path, there is an upwash of air in front of the blade and a downwash behind, the same effect as occurs on an aircraft wing. The net result of this up and downwash is a general downwash of the relative airflow over the blade. Because the total reaction is at a right angle to the relative airflow, it will be tilted rearwards from the vertical relative to the blade element. The horizontal component of the total reaction represents the propeller's induced drag, or to use the modern terminology, trailing vortex drag.

The pressure differential between the propeller blade's top and bottom surfaces causes the air flowing over and under the blade to meet at the trailing edge at an angle to each other, known as the 'rake' and this causes a vortex sheet to emanate from each blade. High aspect ratio blades produce a smaller rake angle and therefore, less induced drag than low aspect ratio blades. This once again reflects the superior efficiency of high aspect ratio propeller blades. The helical vortex sheets flowing off each blade affects the following blade by causing a disturbance in the airflow pattern resulting in a loss in efficiency. The greater the number of blades, the greater is the disturbance. Using more blades negates the advantage of greater solidity due to the increased flow disturbance. This is one reason why six blades on a propeller are the maximum numbers that can be used before efficiency deteriorates beyond an acceptable limit.

The flow around the blade tip from high to low pressure (rear to front surface) causes the tip vortex to be stronger and cause more drag than the trailing edge vortex sheet. A similar vortex emanates from the blade root and this rotates in the same direction as the propeller, while the tip vortex rotates in the opposite direction.

On an aircraft wing, vortex drag can be reduced by using an elliptical wing planform or by wing taper, which has the same affect. However, the planform of a propeller blade has to be designed by calculation and is not necessarily elliptical. This is due to the difference in speed between the blade's root and tip affecting a different amount of air mass in a given time. This problem

is partly alleviated by using scimitar shaped blades as found on new turboprop aircraft. These types of blades are designed with extra chord width around the 50% station (or half way along the blade's length). At low-speed during take-off, more of the thrust is produced in this area of the blade. As the aircraft's speed increases, the major part of the thrust is produced further outboard along the curved span of the blade providing better efficiency at higher speeds where the effects of compressibility are delayed reducing drag and noise. This effect can be likened to a swing-wing fighter aircraft where the wing is swept forward for low-speed flight and swept back for high-speed flight.

The propeller drag is caused not only by the blades, but also to a lesser degree by the prop boss (the thick, rounded, non-thrust producing part of a fixed-pitch propeller). The blade roots and boss cause profile drag due to their inherent thickness. This is one reason for installing a propeller spinner to smooth the airflow over the drag producing areas of the prop. At low-speed ratios, the loss of efficiency caused by drag is around 10% rising to 20% at higher speed ratios. The power required to overcome the profile drag is known as the profile drag power loss. Because aerodynamic forces are proportional to the square of the speed, it would appear obvious the thrust and torque would be at a maximum at the propeller tips where the speed is greatest. However, this is not so. Due to tip losses caused by the spanwise flow along the blade towards the tips and the effects of compressibility, the thrust and torque values reach a maximum around the 80% prop radius station and decrease towards the tip.

*See Feathered propeller; Windmilling propeller forces.*

**Propeller blade face** The relatively flat surface corresponding to the wing's underside is known as the blade face. This is the part of the propeller seen when viewed from the rear.

It is also known as the propeller face or blade face.

**Propeller blade loading** Propeller blade loading is equal to brake horsepower (BHP) divided by the propeller blade area, as opposed to propeller disc loading, which is BHP/propeller disc area.

$$\text{Prop blade loading} = \frac{\text{brake horsepower}}{\text{propeller blade area}} = \text{HP/sq. ft.}$$

Increasing the number of blades can reduce the propeller blade loading for a given prop. It is akin to wing loading where a reduction in prop blade loading (or wing loading) results in the reduced strength of air circulation and a reduction in the velocity of the airflow over the prop blades. The reduced airflow over the blades may at first be puzzling when examining *Diagram 67, Propeller Forces in Cruise Flight*, which shows the relative airflow to be dependent on the RPM and forward velocity.

The air mass on flowing through the propeller disc has an increase in velocity in front of it and a decrease in velocity behind it. With the reduced air circulation over the blades, this increase in velocity is less. The formula for lift (or thrust) can also be used to solve this problem. The formula is:

$$\text{Thrust} = C_L \frac{1}{2} \rho V^2 S.$$

We can cancel out air density ( $\frac{1}{2} \rho$ ) and  $C_L$  (lift coefficient) leaving  $V^2$  and  $S$  as the two variables. Adding more blades to the prop increases the prop blade area ( $S$ ). Therefore, velocity squared ( $V^2$ ) must decrease in order to maintain the same thrust.

**Propeller blade planform** The propeller blade planform is designed so that the chord increases from the root to a maximum value between 50% and 75% radius station and decreases towards the propeller tip. The minimum drag coefficient occurs at the position of maximum chord, with the maximum value of the drag coefficient found at the blade tip due to induced drag and at the blade root due to form drag. The overall drag coefficient will remain approximately constant if the propeller tips are not affected by compressibility.

**Propeller blast** Propeller blast is an alternate name for propwash, the slipstream behind the propeller.

**Propeller boss** The boss is the thick, central, non-aerodynamic part of a wooden or metal fixed-pitch prop. It produces profile drag but no thrust.

**Propeller disc area** The total area swept by the propeller blades is known as the propeller disc area. It is also known as the disc area for short.

**Propeller disc loading** Propeller disc loading is defined as the engine's brake horsepower divided by the prop disc area.

If the propeller's diameter is increased, it will lead to an increase in the propeller disc area. This will reduce the propeller disc loading and in turn, increase the propeller efficiency. Do not confuse the propeller disc loading with the propeller blade loading. The propeller disc loading may be reduced by either increasing the prop diameter or by reducing the engine's brake horsepower. If the engine's horsepower is increased and a propeller of the same diameter is used, it follows the propeller disc loading will also be increased. However, if the increase in propeller disc loading is too great, a loss in efficiency will result. This is due to the increased air pressure at the rear of the propeller disc leaking around the propeller tips causing an increase in propeller tip vortices and induced drag. The same affect occurs on a wing that is too short for the aircraft. In fact, the propeller disc loading is akin to the aircraft's wing loading.

The value of the propeller disc loading can be found given the engine's brake horsepower and the propeller's disc area, from the formula:

$$\text{Propeller disc loading} = \frac{\text{Brake horsepower}}{\text{Prop disc area}}$$

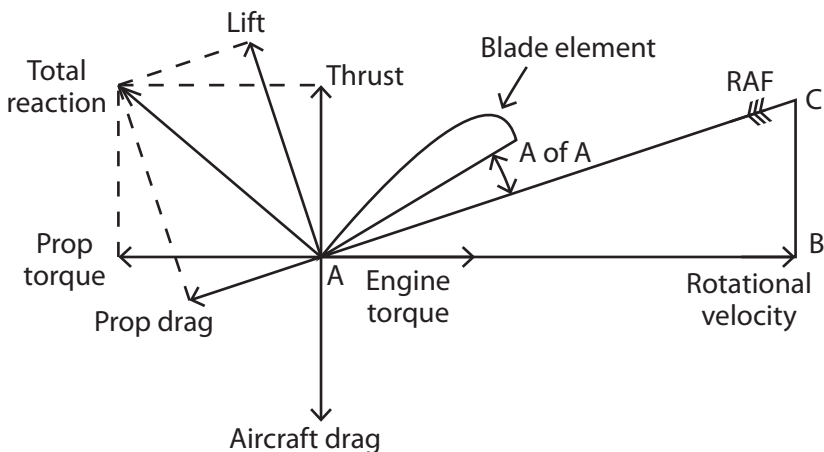
**Propeller efficiency** The purpose of the propeller is to convert the engine torque into axial thrust or slipstream. How well the propeller achieves this is measured by the propeller's efficiency, expressed as a percentage, which is defined as the ratio of propulsive thrust power to engine power, or thrust out divided by power in.

$$\text{Efficiency} = \frac{\text{thrust} \times \text{TAS}}{\text{drag} \times \text{RPM}}$$

This formula can be broken up into two parts to read thrust/drag ratio and true air speed/revs per minute (TAS/RPM) ratio known as the speed ratio. Any change in the propeller's helix angle due to a change in either propeller RPM or true air speed will increase one ratio and decrease the other by a like amount. The thrust is required to be as high as possible, because greater thrust means greater forward speed for a given horsepower. Conversely, drag is required to be as low as possible to allow for an increased forward speed for a given amount of thrust; that is, a high thrust drag ratio is required.

**Propeller forces in cruise flight** *Diagram 67, Propeller Forces in Cruise Flight*, shows the propeller's rotational velocity (A-B) in the propeller's plane of rotation and the aircraft's forward velocity (B-C), which also represents the inflow velocity due to forward motion in the opposite direction (from C to B). The hypotenuse of the triangle (A-C) represents the relative airflow into the propeller disc (from C to A). [Adding the inflow and rotational velocity vectors give the relative airflow vector]. This is determined by the propeller's speed ratio, which is the ratio of the aircraft's forward speed to the propeller RPM. The vector A-C also represents the helical flight path of the blade as it travels from A to C.

Due to the propeller blade's motion along the helical path and its angle of attack, an aerodynamic resultant force is produced; the same effect as found on an aircraft wing in flight. The total reaction (or resultant force) can be divided into the components of lift and drag, but of greater importance to the propeller, into the



**Diagram 67, Propeller Forces in Cruise Flight**

components of thrust and prop torque. The blade's lift coefficient, air density and inflow velocity and blade area govern the strength of the total reaction. This is shown by the familiar lift formula.

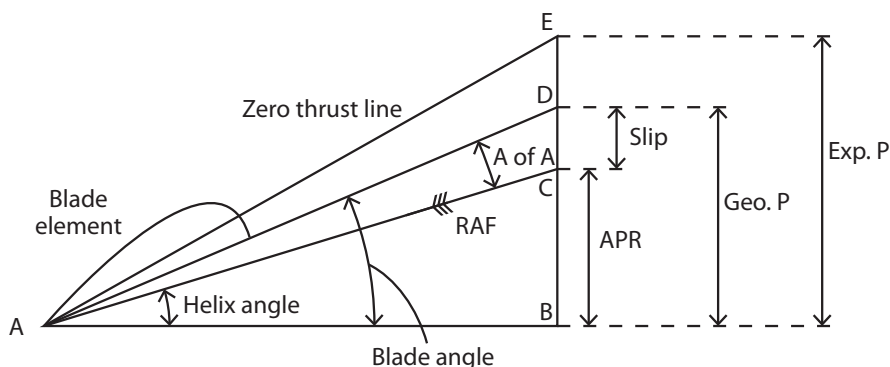
Engine torque acts part way along the vector A-B and the propeller torque acts in opposition to the engine torque. Ignoring the effects of a constant speed unit, the RPM will remain constant as long as these two forces are in opposition. It should be noted here, the propeller's thrust is equal not only to the aircraft's drag but also to the forward component of the total reaction. If the lift component is greater than the thrust component, why doesn't the propeller produce more thrust? This can only be attributed to the fact the forward component of lift, that is, the part greater than the thrust, is equal and opposite to the rearward component of propeller drag. Therefore, these two forces are equal and opposite and cancel out, leaving propeller thrust to equal aircraft's drag.

The vectors of the total reaction, lift, drag, thrust and torque have been drawn as emanating from the trailing edge of the blade element for the purpose of clarity. In practice, this would not be true because the forces all act from the blade's centre of pressure.

*Compare with **Diagram 88, Total Rotor Thrust.***

*See **Normal propeller state.***

**Propeller hub** The hub is in the same central position as the boss but it is a separate unit to which the blades and constant-speed unit is attached.



**Diagram 68, Propeller Pitch**

**Propeller pitch** *Diagram 68, Propeller Pitch*, explains the various terms associated with propeller terminology. It shows the difference between the geometric and experimental pitch, advance per revolution (APR), slip, helix and blade angle, etc. The advance per rev (B-C) plus the slip (C-D) is equal to the geometric pitch (B-D). The advance per rev, also known as the effective pitch, is less than the geometric and experimental pitch under normal operating conditions. The advance per rev is a variable. When the advance per rev is equal to the experimental pitch, the angle of attack of the blade is slightly negative with zero slip and thrust. When the advance per rev is equal to the geometric pitch, the angle of attack is zero and a small amount of slip and thrust are still present.

See *Diagram 67, Propeller Forces in Cruise Flight*.

**Propeller region** A reference to a helicopter's main rotor disc.

See **Driven region**.



**Propeller shank** The propeller shank is located at the blade root. Being circular in form it is not a lift or thrust producing shape and therefore plays no part in producing thrust, although it does produce some drag.

**Propeller slip** Propeller slip is defined as 'the difference between the advance per rev and the geometric pitch'.

Maximum efficiency is only obtained when the slip, expressed as a percentage, is around 30% of the length of the geometric pitch (B-D). In other words, slip (C-D) shown on **Diagram 68, Propeller Pitch**, is the difference between the actual distance the propeller travels forward in one revolution (B-C) or advance per rev and the distance the propeller would theoretically travel in one revolution if its advance per rev was equal to the geometric pitch (B-D). That is, slip is a percentage of distance only. Slip is related to the geometric pitch and thrust.

See **Diagram 67, Propeller Forces in Cruise Flight**.

**Propeller slipstream** The propeller slipstream is correctly referred to as propeller wash or propeller blast (propwash or prop blast for short). However, the more commonly used term is slipstream.

Propeller powered aircraft have the added advantage of the slipstream flowing over that portion of the wing behind the propeller. The increased slipstream velocity acts in the form of a boundary layer control and increases the lift coefficient and the amount of lift produced by that part of the wing. Due to this extra lift, the engine power should be reduced gradually when in the flare for landing, otherwise the sudden loss of slipstream velocity and the lift it produces can result in the plane dropping to a heavy landing. Jet aircraft are of course, devoid of any propeller slipstream and its advantage found on propeller aircraft.

See **Helical slipstream**.

**Propeller spinner** A spinner is the streamline fairing covering the boss or hub area of a propeller. It may be used to smooth the airflow into the engine intakes, reduce drag around the blade root area or for aesthetic reasons.

**Propeller stability** The location of the prop/engine affects the aircraft's stability due to its position and the presence of the slipstream.

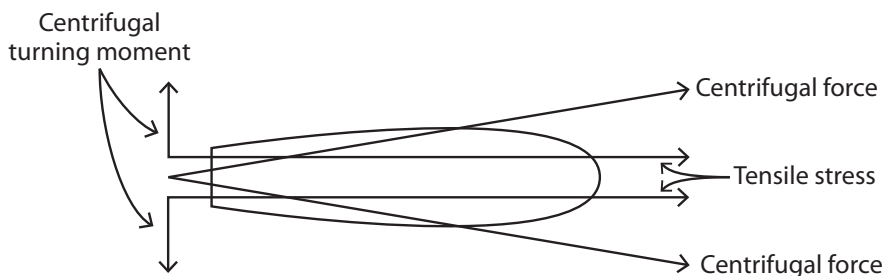
Although not as common as tractor propellers, rear mounted pusher props enhance the stability. If the aircraft experiences a yaw say to the left, the slipstream will be deflected the opposite way, to the right and will pull the aircraft's nose back around to the right aiding directional stability. Imagine the slipstream acting in the same sense as the rudder. The same effect occurs in the pitching plane with the slipstream bending in the appropriate direction to raise or lower the nose, acting like the elevator. Both actions stabilize the aircraft.

Conversely, the tractor prop can be de-stabilizing. If an aircraft with a tractor prop is induced to yaw to the left (as in the example above) the slipstream will still deflect to the right, but will pull the aircraft's nose further to the left; this results in a de-stabilizing of the static directional stability. In a yaw condition the propeller acts like a fin mounted on the nose of the aircraft forward of the centre of gravity. Single-engine aircraft are more prone to this problem with its engine mounted further forward of the CG than a twin-engine aircraft with wing mounted engines. In addition, the same de-stabilizing factors occur in the pitching plane.

**Propeller stress** Stress can be divided into four main categories of torsional stress, tensile stress, compression, and bending stress. Any one stress or combinations of stress can be present acting on an aircraft or propeller at any one time. Stress is measured as a force per unit area. All aircraft are subject to stress and must be flown within the operating envelope to ensure the stress limitations are not exceeded. This applies to the airframe, engine and propeller. Propeller stress is a good example of the different types of stress, which are used here as an example.

Stress occurs whenever a load is applied, such as is found with a rotating propeller. The stress will vary depending on the operating conditions of the propeller; the stress force can be reversed if the propeller is windmilling or with reverse thrust application. Life for a propeller is not an easy one!

Reference to **Diagram 69, Blade Stress Forces**, shows the centrifugal force caused by the spinning propeller acting from the propeller hub through the leading and trailing edges of the propeller blade. The centrifugal force is divided into two components, one causing tension (tensile stress) due to stretching and the other component causing a centrifugal turning moment (torsional stress).



**Diagram 69, Blade Stress Forces**

The tensile stress tends to flow through the propeller blades outwards, in effect stretching the blade in length and generating a considerable force on the propeller hub and increases towards the propeller tip.

The centrifugal turning moment (torsional stress) acting at 90 degrees to the propeller axis produces a turning moment around the propeller pitch change axis tending to turn the blades towards fine pitch. This force is opposed by the pitch change mechanism, which is placed under stress, but is assisted to a certain degree by the aerodynamic turning moment during normal cruise conditions that, tends to turn the blades towards coarse pitch. The pitch change axis is normally to the rear of the blade's centre of pressure. The resultant aerodynamic force acting at the centre of pressure will tend to turn the propeller blades in the opposite direction; the direction here being reversed when the propeller is windmilling causing the resultant force to act rearwards. This time the aerodynamic turning moment is acting in the same direction as the centrifugal turning moment and turns the blades towards fine pitch. The centrifugal turning moment is a more powerful force than the aerodynamic turning moment and causes the most stress. Torsional stress forces increase with the RPM squared.

Additional to the stress mentioned above, is the bending stress on the blades caused by the thrust (blade loading) of the propeller. The blade loading is measured in horsepower per square foot. The force generated by the propeller tends to bend the blades forward when under power, while the tension in the blade caused by the centrifugal force opposes the thrust, which tends to straighten the blades out. Bending stress will be reversed in direction with reverse thrust application on the landing rollout.

**Propeller tip speed** The propeller blades on light aircraft are limited to 72 to 76 inches diameter with engines operating at around 2400 RPM cruise speed. Turbo-prop aircraft have prop diameters twice this size but they are geared to run at a much lower RPM. Too high a tip speed and the tips will run at transonic speeds and will lose their efficiency and cause excessive noise.

The propeller tip speed can be found in feet per second given the RPM and diameter from the following formula:

$$\text{Prop tip speed (FPS)} = 2 \pi R N$$

Where  $\pi = 3.14$

R = prop radius in feet

N = prop revs per second.

Two aerodynamicists from the Royal Aircraft Establishment (Farnborough), R. Wood and G. P. Douglas, conducted propeller research in the low-speed wind tunnel at the National Physical Laboratory, near London, UK. A DeHavilland DH.9A biplane was used in flight-tests to determine if the propeller's thrust, torque and prop tip speed were too high, causing compressibility at the tips.

Propeller tip speed is also known as the tangential velocity.

**Propeller theories** Several theories have been proposed on how propellers produce thrust. The theories include the following:

- vortex
- general momentum
- axial momentum
- blade element.

The blade element theory is the most widely accepted theory along with parts of the axial momentum theory, leaving the vortex and general momentum theories to fall by the wayside.

**Propeller torque force** In providing power to turn the propeller, the engine produces a torque component, which is force acting in the same plane and direction as the propeller rotation. At a constant power setting, the engine torque is balanced by the equal and opposite force of propeller torque, in accordance with Newton's third law. On an aircraft with a fixed-pitch propeller, the engine RPM will remain constant as long as these two forces remain in balance. A change of power setting or aircraft speed will change the value of the engine torque or propeller torque respectively. This will result in a change in engine RPM. The same applies to a constant-speed propeller but the constant-speed unit works to maintain a constant RPM. The torque varies as the RPM squared so by doubling the RPM from say, 1200 to 2400 RPM on a static ground run-up; the torque will increase by four. The total drag force or torque of the propeller acts through the centre of pressure of each propeller blade. The propeller torque can be found from the following formula:

$$\text{Prop torque} = N (C_D V^2 S) R$$

Where N = number of prop blades

$C_D$  = drag coefficient

$V^2$  = prop tip speed squared

S = blade area

R = prop radius

Note:  $V^2 = (2 \pi R N / 60)^2$  where N = RPM.

See **Propeller tip speed; Torque roll and stall.**

**Propfans** In 1980, Hamilton Standard introduced a proof of concept advanced turboprop engine designed for short to medium haul airliners. The new turboprop engine was designed to power a propeller of advanced design, with five to thirteen swept-back scimitar shaped blades, having a critical Mach number of 0.8 to 0.83. In 1976, NASA's Lewis Research Centre in Cleveland, Ohio, contracted to Hamilton Standard to jointly research and develop an advanced turboprop engine driving a propeller with eight titanium blades to be efficient at a cruise speed of Mach 0.8. Titanium was chosen because normal metal blades are too thick and therefore inefficient at the transonic tip speeds the propeller was designed to run at. This propeller became known as the Propfan, a term now generally used to describe such propellers.

NASA later joined forces with General Electric (makers of jet engines) to produce a Propfan using composite blades for their strength and lightweight. General Electric copyrighted the name of their Propfan as 'Unducted Fan' (UDF®). The first flight of the GE-36 UDF occurred on 20 August 1986 at Edwards AFB, California and testing ended in February 1987. The engine was attached to the right-hand engine mount of a Boeing 807-100 test plane to become the first Propfan to fly. The Unducted pusher Propfan had an eight blade contra-rotating propeller of 11ft (3.35 m) diameter.

General Electric also flight-tested a Propfan on a McDonnell Douglas MD-80 between May 1987 and March 1988. This Propfan also had eight scimitar shaped contra-rotating blades, driven by the GE turbine engine with an un-gearred direct drive to the unducted fan. This resulted in a greatly improved performance over a conventional jet engine. McDonnell Douglas called their Propfan a 'UHB' (or an ultra-high by-pass ratio engine) with scimitar blades. It produced a by-pass ratio of 36:1 compared to the 5:1 ratio of a Boeing 747's JT5D engines. After four years of research and development at a total cost of around \$100 million, McDonnell Douglas cancelled the project due to the high cost of the engine.

Pratt & Whitney joined forces with Allison to test the P&W Allison Model 578-DX 20000 lb. static thrust turboprop with a Hamilton Standard six-blade contra-Propfan of 11 feet 7 inches (3.53 m). The first flight of the MD-80 with the geared Propfan took place on the 13 April 1989, with all testing being completed in a few weeks.

It appears the big American companies are no longer involved in any Propfan projects. In the Commonwealth of Independent States (formerly Russia) research continues. An Ilyushin IL-76 test bed aircraft was used to test the Lotarev contra-rotating tractor Propfan mounted on its number two (port inner) engine mount, with the first flight being made on 25 March 1971. The YAK-46, a 168-seat airliner with a pusher Propfan, is powered by two 24690-pound static thrust ZMDB Progress D-27 Propfans with a



**Photo:** The Hamilton-Standard HS 12-blade contra-Propfan is the ultimate in propeller technology. The Propfan is shown here mounted on a McDonnell Douglas MD-80 UHB demo test plane and is powered by a P&W/Allison 578-DX turbine engine. It never entered production.

*Photo: Supplied courtesy of Hamilton-Standard of Connecticut, USA.*



first flight conducted around 1995. It never went into production. The Antonov AN-180 was another new type of passenger aircraft on the drawing boards around this time. It was due to fly around 1995 and was powered by two rear-mounted tractor Propfan engines the same as found on the YAK-46.

The Antonov AN-70 was designed as a military and civilian large STOL transport with a maximum all up weight of 286 600 lb (130 000 kg). Built in the Ukraine, the prototype made its first flight on 16 December 1994 but its life was short lived, being lost in a mid-air collision with its chase plane on 10 February 1995. The second prototype first flew on 24 April 1997 with a third under construction in 1998. The four-tractor Progress/Motor Sich D-17 turboprop engines of 14000SHP each, driving contra-rotating Propfans, which are different to the usual Propfan configuration. The Propfan has eight blades mounted on the front prop and six blades mounted on the rear assembly. The Antonov AN-70 is now in production with models going to the Russian, Ukraine and Czech Republic Air Forces and a civilian model is available.

It now appears that Russia is further advanced than the USA in Propfan technology and the use of Propfan powered aircraft.

**Prop-rotor** This is an alternate term for a tilt-rotor system, where the engine and its prop rotate through 90° to provide VTOL manoeuvres and horizontal flight.

It is also known as a tilt-prop.

**Propulsive efficiency** The efficiency of the aircraft's propulsive system is determined by the energy imparted to the aircraft, being a percentage of the energy produced by the propeller or jet engine. Propulsive efficiency should not be confused with propeller efficiency. A jet engine's propulsive power available is the product of the aircraft velocity and its engine thrust available. Therefore, the propulsive efficiency is greater when the jet engine's exhaust velocity or propeller thrust is at a speed coincident with or nearly so, to the aircraft's velocity. Propulsive efficiency is expressed by:

$$Pe = \frac{\text{propulsive power developed}}{\text{rate of increase in kinetic energy}}$$

$$= \frac{TV}{T(V + a)}$$

Where T = thrust, lbs

V = true air speeding Knots

a = accelerated flow through propeller disc.

Different engine types and propellers are suited to the design cruise speed of the aircraft. At the low end of the speed range, the piston-engine powered propulsor is ideal where cruise speeds are about 80 Knots maximum. Next in line is the piston-engine/propeller combination suitable for speeds up to about 200 Knots or so. Above that speed, turboprop engines are more suited for speeds up to about 330 Knots; Propfans and turbofan engines used on jet transports produce their peak efficiency up to about 650 Knots. Supersonic jet fighters use turbojet engines, which are good for speeds up to about 1500 Knots, where rocket and ramjet engines must take over. Therefore, propulsive efficiency can be used to compare different types of engines.

*See Ideal efficiency.*

**Propulsive power** *See Power.*

**Propulsive thrust** The net propulsive thrust is the force driving the aircraft after subtracting the various losses from the free thrust.

*See Thrust terminology.*

**Propulsor** Unlike the Propfan, which is a relatively new idea, the propulsor has been around for many years. A propulsor is simply a propeller mounted inside a shroud. Although it has several advantages over conventional un-shrouded propellers, it has never been greatly utilized by aircraft designers because of its suitability only for low-speed operations.

The propulsor can trace its origin back as far as 1910, to the Bertrand Monoplane of French design. It had a single-engine driving a tractor and pusher propeller, one at each end of the shroud. This was followed in 1932 by the Italian built Stipa-Caproni, with a 120 BHP DeHavilland Gypsy III engine and propeller, which were both, mounted inside the shroud. A more recent example of a propulsor-equipped aircraft appeared in 1966, in the form of the experimental Bell X-22A with four turbine engines driving four separate tilt-vector propellers mounted inside shrouds.

The US/German, VFW Rhein Flugzeugbau/Grumman American joint venture Fanliner with two seats and a pusher propulsor was another example. The Brooklands/Edgeley Optika from the UK, designed for the observation role, is one of the more recent examples to be built. It was showing good prospects for being a commercial success when production came to a sudden end. The factory was destroyed by fire and the production line was never re-established. However, prior to this event, the prototype's first flight was on 14 December 1979. A 200/210 BHP Lycoming engine, giving the aircraft a cruise speed of 57–108 knots powers the five-blade fixed-pitch propeller. Due to the propulsor, it is said to be the world's quietest aircraft.

Present day airships, such as those built by Airship Industries of the UK, have relatively low cruising speeds and are ideally suited to the propulsor power unit.

**Propwash** The slipstream immediately behind the propeller. The alternate and correct name for propeller-blast.

**Propwash-thrust** The Rankine-Froude axial momentum theory gives a clear explanation of the energy change in the propeller's propwash, known as the propwash-thrust; the term slipstream-thrust is incorrect.

The marine engineer Robert Edmund Froude introduced the idea of the propeller disc, later known as Froude's actuator disc, where the air mass on passing through the propeller disc experiences a sudden rise in pressure without affecting the increasing propwash velocity. The axial momentum theory assumes incorrectly, the propeller thrust is evenly distributed across the propeller disc, which is not true. In fact, the thrust varies from zero at the hub to a maximum at the 0.80 radius station and reduces again to zero at the propeller tips. No propeller achieves the efficiency an actuator disc implies; however, this fact will be disregarded for now and it will be assumed the thrust to be evenly distributed across the propeller disc.

As the propeller rotates under normal operating conditions, it sucks air from in front of the propeller disc causing a low-pressure area. The air mass known as the inflow velocity ( $V_i$ ) accelerates through the propeller disc into an area of increasing velocity behind the propeller and experiences a rapid rise in pressure as it does so. It is the difference in pressure between the front and rear of the propeller disc, which is caused by the change in the air mass momentum that produces thrust. The air mass now called the outflow velocity ( $V_o$ ) continues to accelerate reaching its maximum velocity some distance behind the propeller. The final maximum velocity is equal to the aircraft's true air speed plus double the propwash velocity (aircraft speed +  $2V$ ). It should now be obvious from the above, half of the propwash velocity increase occurs in front of the propeller while the other half of the speed increase occurs behind the propeller. As the speed of the propwash increases to its maximum value, the slipstream also contracts to a

smaller diameter than the disc itself, in compliance with Bernoulli's Theorem. At the point of constriction, it is known as the vena contracta. The air mass flowing through the propeller disc should be considered as a three dimensional stream tube.

The propwash velocity/aircraft velocity ( $v/V$ ) ratio is known as the inflow factor, which increases with an increase of propwash velocity. Propeller efficiency will be greatest when the propwash velocity is close to the aircraft's true air speed ( $V$ ) that is, greatest efficiency is achieved at a small value of  $v/V$ . When the propwash velocity equals the aircraft's velocity, the value of the inflow factor is equal to one. At other speeds, where the propwash velocity is less than the aircraft's velocity (which is the usual case) the value of the inflow factor is of course, less than one. This ties in with the 'ideal efficiency' of the propeller, where the thrust is related to the aircraft's speed and the inflow factor. The ideal efficiency, or Froude's efficiency, can be calculated from the formula known as Froude's Equation:

$$\text{Ideal efficiency} = \frac{T \times V}{T \times (V + v)}$$

Where  $T$  = thrust factor

$V$  = aircraft's true air speed

$v$  = propwash velocity.

As the aircraft accelerates from rest, the propwash velocity will also accelerate but not as quickly, until a point is reached where the aircraft's velocity and the propwash velocity are both identical. This occurs at the maximum level-flight speed of the aircraft. Stated simply, thrust equals the mass of the air multiplied by the propwash velocity minus the aircraft's velocity. As the aircraft travels forward, the air mass well ahead of the propeller disc is assumed stationary. On approaching the propeller disc and passing through it, the air mass accelerates and gains momentum. Thrust is the result of this change in momentum and the greater the change, the greater the thrust.

**See Axial momentum theory.**

**Puller** A tail rotor mounted on a helicopter is called a puller if it pulls the tail boom to counteract main rotor torque. Many early helicopters had puller anti-torque rotors; modern helicopters have pusher tail rotors.

On airplanes, the term puller does not apply to the propeller; instead it is called a tractor propeller.

**Pusher prop** A pusher propeller is mounted aft of the engine, which pushes the aircraft along, hence the name 'pusher', as opposed to 'tractor' props, which pulls the aircraft forward. However, to be precise, propellers neither push nor pull! It is the difference in air pressure in front of and behind the propeller (the change in airflow energy), which provides the aerodynamic reaction to propel the aircraft forwards. The pusher and tractor props both work in the same sense.



**Photo:** The Convair B-36J Peacemaker must be the largest aircraft with pusher props. It has six of them no less, plus four jet engines! The photo shows the last B-36 to be built, and it is homed at the Pima Air & Space Museum, Tucson, Ar.

The accepted terms of 'tractor' and 'pusher' props are used to describe the location of the propeller on the aircraft. Compared to tractor propellers (the most common configuration), pusher props have several advantages and disadvantages; some are listed below.

The advantages of pusher props are:

- re-energized boundary layer over the wing or fuselage by maintaining an attached air flow forward of the propeller
- profile and form drag is reduced
- greater inherent stability requires less fin area
- reduce affect of cross-wind on take-off or landing
- a shorter fuselage, reducing weight and wetted area
- lack of rotary airflow from the propeller inducing a side-force on the fuselage.

The disadvantages are:

- the aft centre of gravity range is reduced due to the location of the aft CG limit (the airplane is more prone to flat spins)
- prop efficiency is reduce by 2–15% due to disturbed airflow over wings or fuselage or downwash from trailing edge of wing flows through prop disc at an angle
- pitch/power coupling may be increased with changes in propwash over tail with engine power changes, increasing pilot work load
- wing leading edge ice breaking off can damage the prop blades or be deflected into the side of the fuselage
- pusher engine/props mounted on the wing trailing edge take up space normally occupied by flying control surfaces
- noisier due to the engine exhaust striking the propeller blades, more so with turboprop engines with their greater power and exhaust.

A helicopter's tail rotor is also termed a pusher if it pushes the tail boom to counteract main rotor torque. This is the most common installation on modern helicopters as opposed to 'pullers' on earlier models.

*Diagram 67, Propeller Forces in Cruise Flight*, describes the forces acting on the prop with no distinction between pusher and tractor props.

*See Propeller aerodynamics; Propeller stability.*

**Push/pull configuration** The push/pull configuration refers to the placement of the engine/propeller combination on the aircraft.



**Photo:** The twin-engine Cessna P337 Skymaster has an engine mounted at each end of the fuselage pod.

Claudius Honoré Dornier (1884–1969) a German airplane designer and manufacturer patented his design for the push/pull configuration in 1937. His 1940s design of the Dornier Do 335 Pheil (Arrow) heavy fighter had an engine at each end of the fuselage. One of his earlier designs, the giant Dornier Do X flying boat had twelve push and pull engines mounted in tandem on the wings. Cessna used the push/pull configuration on their twin-engine Cessna 336/337 Skymaster, with an engine mounted at each end of the fuselage pod.

It is more commonly known as a centreline thrust configuration.

**Pylon whirl** An unfavourable aeroelastic, dynamic, rotor instability of a helicopter's rotor blades pylon.



# Q

**‘q’** The letter ‘q’ (lower case) is an abbreviation for dynamic pressure ( $\frac{1}{2} \rho V^2$ ) due to the forward motion of air in an incompressible flow that is brought to rest. The dynamic pressure is measured in multiples of ‘q’. It is the difference between total and static pressure.

This is a more convenient value to calculate lift than using pounds per square inch or similar units.

**Q-feel** Modern jets typically have powered flight controls making it easier for the pilot to move them. However, powered controls do not have any feel to them (stick force per ‘g’) and so an artificial feel is built into the control system. The feel varies with the amount of control deflection selected and with air speed, or to be precise with dynamic pressure. Dynamic pressure is quoted as  $\frac{1}{2} \rho V^2$  or as ‘q’, hence the term Q-feel.

See **Stick force per ‘g’**.

**‘Q-tip’ propellers** Q-tip propellers blades have a type of end plate or bent-up tip similar to winglets found on modern jet aircraft wings. They are there to reduce backside pressure leaking around the tips and enable higher propeller blade span loadings to be achieved at a lower RPM than normal, thus reducing propeller noise and improving efficiency. The reduced diameter produces lower propeller tip speed, which also helps to reduce noise for the same amount of thrust.

Hartzell Propellers in the USA introduced the Q-tip propeller and the first aircraft to use them was the Piper PA-42 Cheyenne.

**Quarter chord point (c/4)** On most general aviation airfoils, the front quarter chord area of the wing provides about half the lifting force. This is clearly shown by the **Diagram 53, Lift Distribution**. With the majority of the wing’s lift in this area, the pitching moment with angle of attack changes is reduced to an acceptable level. The main wing spar is usually located at or near the quarter chord point for structural reasons associated to the wing lift.

# R

**Race rotation** This is the rotational velocity imparted to the slipstream by the propeller.

See **Helical velocity**.

**Radius of turn** The term radius of turn is commonly used in aerodynamic textbooks, which is only of academic interest to pilots. It is the diameter of the turn that pilots use in practical flying. The position of the aircraft after completing a turn onto a reciprocal heading will be equal to the diameter of the turn – twice the radial distance.

The radius of the turn is proportional to the true air speed squared and inversely proportional to the angle of bank. Therefore, the higher the true air speed, the greater is the radius of turn, at a given rate of turn. Conversely, the greater the angle of bank, the less is the radius of turn, at a given true air speed. The aircraft's weight does not come into the equation, which applies equally to all aircraft.

The radius of turn can be found from either of the two formulas below, using Knots true air speed (KTAS) divided by the product of the factor 11.26 and the angle of bank ( $\tan \theta$ ), or alternatively by dividing the TAS squared by the acceleration 'g' multiplied by angle of bank, thus:

$$R = \frac{V^2}{11.26 \tan \theta}$$

$$\text{or, } R = \frac{V^2}{g \tan \theta}$$

Where R = radius of turn, in feet

V<sup>2</sup> = Knots true air speed squared

11.26 = constant or,

g = 32.2

$\tan \theta$  = angle of bank in degrees.

A simple rule of thumb to find the radius of turn in nautical miles is to take 1% of the ground speed in Knots, (this distance must be doubled to find the diameter of turn, which is of more practical use to pilots).

*See* **Load factor; Rate of turn; Radius of turn; Diameter of turn; Turning.**

**Rake angle** The pressure differential between the wing's top and bottom surfaces causes the air flowing over and under the wing to meet at the trailing edge at an angle to each other, known as the rake angle. The airflow over the upper surface flows chordwise with an inboard component. The lower surface airflow tends to have an outboard flow (although, some wings may have an inboard flow but not as great as the upper surface flow). Either way, the two flows meet at the trailing edge with a rake angle.

The airflow causes a vortex sheet to emanate from each wing. High aspect ratio wings produce a smaller rake angle and therefore, less induced drag than low aspect ratio wings, reflecting the superior efficiency of high aspect ratio. The flow around the wingtip from high to low pressure (bottom to top surface) causes the tip vortex to be stronger and cause more drag than the trailing edge vortex sheet. Vortex drag can be reduced by using an elliptical wing planform or by wing taper, which has the same affect.

*See* **Aspect ratio.**

**Raked wingtips** Raked wingtips are a form of wingtip device used on some large jet transport aircraft.

Boeing Aircraft favors the raked wingtip over the more conventional winglets. As the name suggests, the raked wingtip has a greater angle of sweepback than the wing's leading edge. It produces a similar effect to the winglet by increasing the effective aspect ratio, which in turn reduces the induced drag and tip vortex. An increase in fuel efficiency, climb performance, and reduced take-off distance are all benefits of raked wingtips.

NASA and Boeing's joint research showed that drag was reduced by up to 5.5% more than that saved by conventional winglets. Adding raked wingtips increases the wingspan, which in some planforms is preferred over winglets of shorter length for long-range routes. Boeing 747-8, B777-200LR and 300ER and the new Boeing 787 all use raked wingtips except the short-range B787-3 model. However, the extra length of wingspan may not be acceptable at some airports due to insufficient clearance at the gate.

*See Wingtip devices.*

**Ram effect/recovery** Initially, an increase in aircraft forward speed will cause the intake mass air flow velocity to decelerate and back-up in front of the compressor causing intake momentum drag (ram drag), which increases with the square of the air speed. With increasing aircraft forward speed, the ram drag rise is overcome by ram recovery effect and engine performance increases.

Ram effect (or ram recovery) is an advantage for engines where the increased air pressure and density helps to produce greater engine power due to the increased air mass per unit volume entering the engine. This occurs at a speed of around 300 Knots. Ram recovery or ram rise, as it is called when referring to jet engines, is achieved due to the inlet air duct being of a divergent shape. The intake air velocity is reduced, which increases the pressure energy (ram air compression) and smoothes the intake airflow into the engine fan. The effective design of the engine intake recovers the losses due to intake momentum drag, and the engine performance improves.

Ram effect also occurs with increasing air speeds in Pitot tubes and other air inlets. The disadvantage of ram effect in the Pitot tube is due to the increased air pressure and density causing the air speed indicator to over read; depending on altitude and true air speed, a given value must be deducted from the equivalent air speed (as indicated on the air speed indicator) to find the true air speed of the aircraft.

It is also known as pressure recovery.

*See Intake momentum drag (Ram drag); Convergent/divergent duct; Divergent intake duct.*

**Ram drag** See Momentum drag.

**Rams horn vortex** All wings produce wingtip vortices in flight. However, the tip vortices on sweptback wings are intensified by the formation of a ram's horn vortex.

The ram's horn vortices are formed when the boundary layer flow spreads towards the wingtips, thickens and separates. Starting on the wing's leading edge inboard from the tips, the ram's horn vortex spreads in an arc towards and across the wing to combine with and intensify the wingtip vortex.

Ram's horn vortices are more prominent on high-speed jet fighter aircraft with their highly sweptback wings with a sharp leading edge. The combined rams horn vortex and trailing edge vortex cause a much stronger tip vortex and hence, greater downwash and thus greater induced drag. Fences or notches are used to reduce the formation of ram's horn vortices.

See Sweptback wings.

**Range speed** Range speed is defined as 'the least amount of fuel used per unit distance' in nautical miles.

*Diagram 60, Thrust & THP Required & Available Curves*, shows  $V_Y$  as the maximum rate of climb speed. It is also the maximum lift/drag ratio and range speed. The range speed may be the best speed for range as far as the airframe is concerned, but the engine must also be taken into consideration. The fuel flow and air speed both increase at a variable rate until a given speed is reached where the engine is burning the least extra fuel above optimum. Cruising at the maximum lift/drag ratio speed ( $V_Y$ ) would not be very practical or convenient – planes are meant to fly fast. Therefore, the optimum cruise speed is about one third times greater than the actual range speed (1.32 times the range speed to be exact) on the front side of the power curve at about 65–80% brake horsepower. The range speed can be considered as the theoretical cruise speed and the optimum cruise speed as the practical speed.

The maximum lift/drag ratio speed times 1.32 is known as the Carson speed after its originator, Dr. Bernard H. 'Bud' Carson PhD (1934–2009).

**Rankine-Froude theory** The Rankine-Froude theory is applied to aircraft propellers to describe their method of producing thrust to propel the aircraft, known also as the axial momentum theory. It is attributed to the work of William J. M. Rankine (1820–1872) and William Froude (1810–1879).

*See Axial momentum theory; Propwash-thrust.*

**Rankine-Hugoniot equation** The Rankine-Hugoniot equation is the main equation dealing with the air density and pressure changes through a shockwave in supersonic flow. William J. M. Rankine (1820–1872) presented the theory as early as 1870. In 1881, the Frenchman, Pierre Hugoniot (1851–1887) a ballistician, rediscovered the same equation unaware of William Rankine's research. Professor Ludwig Prandtl further extended this knowledge with one of his students, Theodore Meyer in 1908 at Gottingen, Germany after their research on oblique shockwaves.

*See Shockwaves.*

**Rankine, W. J. M** In 1865, the Scottish engineer and scientist, William John Macquorn Rankine (1820–1872) founded the propeller's axial momentum theory while working on the theory of ship's propellers. Another engineer, R. E. Froude (c1830–?) covered further work and development on the theory. Amongst many other scientific achievements, William Rankine was also one of the founders of thermodynamics, specializing in steam engine theory. He also introduced his Rankine temperature scale where the absolute minimum temperature is recorded as minus 459.67R (-273.15°C) with the freezing point of fresh water as 491.67R (0°C). His scale is based on each degree of temperature being the same value as one degree Fahrenheit, similar to the Kelvin temperature scale. The capital letter 'R' follows the temperature without a degree sign.

*See Rankine-Froude theory.*

**Rapid incidence adjustment**    *See* **Short-term (or short-period) oscillation.**

**Rarefaction** A supersonic airflow passing through a converging duct undergoes a linear decrease in air pressure and air density with an increase in air velocity; this is known as a rarefaction.

*See* **Expansion flow & fan.**

**Rated static thrust**    *See* **Static thrust.**

**Rate of climb** The rate of climb is a measure of the vertical component of the aircraft's flight path expressed as feet per minute (FPM).

The maximum rate of climb is achieved when the engine is delivering the most excess thrust horsepower (ETHP), that is, when the thrust differential between the thrust horsepower required and thrust horsepower available is at a maximum. When the excess thrust horsepower has reduced to zero with full power selected, the aircraft will be at its absolute aerodynamic ceiling. The rate of climb is related to the excess thrust horsepower, as opposed to the angle of climb, which is related to the excess thrust in pounds. The maximum rate of climb can be found from the following formula:

$$\text{Maximum rate of climb (FPM)} = \frac{\text{ETHP} \times 33000}{\text{Aircraft weight}}$$

*See* **Piston-engine power curves – airplane.**

**Rate of descent** The rate of descent of the aircraft is measured in feet per minute; this depends on the deficit thrust horsepower and aircraft weight, as shown by the formula below. Thrust horsepower is required by the aircraft to maintain straight and level flight. Deficit thrust horsepower is the power that is unavailable, for example, when the throttle is closed. The rate of descent can be varied with power available until the power available is zero

at which point the rate of descent will be as given by the formula below:

$$\text{Rate of descent (FPM)} = \frac{\text{Deficit THP} \times 33000}{\text{Aircraft weight}}$$

**Diagram 17, Deficit Thrust Horsepower v. KTAS**, shows the maximum rate of descent is achieved at the same speed as the maximum rate of climb speed  $V_Y$ . The graph shows the thrust horsepower available is located below the thrust horsepower required curve; this is the opposite condition to when the engine is developing power. The difference between the two is the deficit thrust horsepower.

See **Diagram 17, Deficit Thrust Horsepower v. KTAS**.

**Rate of turn** The rate and radius of turn are closely related to the aircraft's angle of bank and true air speed. A steady coordinated turn of three degrees per second is classed as a standard rate turn performed at approximately 15–20 degrees angle of bank, depending on air speed. Flying at a constant rate of turn, the turn radius is doubled when the speed is doubled and conversely, at a constant speed and double the rate of turn, the radius of turn is halved. The rate of turn is independent of aircraft type and weight.

The rate of turn (RoT) can be calculated from the following formula:

$$\text{RoT} = \frac{1091 \tan \theta}{\text{KTAS}}$$

Where: RoT = rate of turn in degrees per second

1091 = constant

KTAS = true air speed in Knots

$\tan \theta$  = angle of bank in degrees.

See **Angle of bank ( $\phi$ ); Diameter of turn; Minimum radius of turn; Radius of turn; Turning**.



**Rate 1 turn** *See Standard rate turn.*

**Reaction torque** A helicopter in the hover can turn about the OZ vertical axis controlled by the tail rotor thrust. Increasing the tail rotor thrust will turn the helicopter in one direction, while decreasing the tail rotor thrust will turn the helicopter in the opposite direction; this is due to reaction torque.

*See Tail rotor.*

**Real gas effects** In hypersonic flows, the changes in the gas (airflow over the aircraft) must be taken into consideration. Temperature, pressure, velocity and the adiabatic index are all relevant to a moving gas at low velocity. However, the real gas effects of hypersonic flows include other variables of dissociation, separation, ionization, and rarefaction due to friction caused by heat.

*See Hypersonic aerodynamics.*

**Rear fuselage form drag** The aft fuselage of most large aircraft is not cone shaped, but curves upwards towards the tailplane. The curve is there to ensure the fuselage does not scrape on the runway during over-rotation on take-off and landing; on cargo aircraft, the area may be occupied by the rear cargo door/ramp.

Too much upsweep can induce an adverse pressure gradient, which causes a portion of the three-dimensional airflow to move towards the aircraft centreline resulting in boundary layer separation and rear fuselage form drag. The disturbed airflow also induces oscillations in the yawing plane (tail wagging). The cure is to strategically place vortex generators to modify and energize the boundary layer flow, in order to reduce the rear fuselage drag.

**Rear loading** A wing is rear loaded when the centre of pressure lies further aft on the upper surface than its normal location of about 25% MAC. Supercritical wings are rear loaded at about 50% MAC.

*See* **Supercritical airfoil.**

**Rearplane** The rearplane is the aft wing on a tandem wing aircraft. To qualify as a rearplane, the two wings must be of similar size; biplanes and canards do not conform. The Quickie 2 is one example of a homebuilt aircraft with a rearplane.

**Rear stagnation point** At the start of the take-off run, a rear stagnation point develops initially on the wing's upper surface forward of the trailing edge, and then moves aft to the trailing edge as the aircraft accelerates.

*See* **Starting vortex; Stagnation point/line.**

**Rebated leading edge** A rebated leading edge is found mostly on jet fighter aircraft sweptback wings. However, it may also be found on other aircraft types. It consists of a notch where the outboard part of the wing's leading edge protrudes further forward than the inboard edge.

*See* **Dog tooth; Notch.**

**Recirculation** The definition of recirculation can be applied to fixed-wing aircraft and to helicopters.

Fixed-wing aircraft have recirculation on the wings when the adverse pressure gradient airflow reverses direction – recirculates. A helicopter in the hover can be affected by recirculation of the airflow through the rotor disc.

The recirculation is self-induced by the helicopter, especially near the rotor tips. The downward flow through the disc is deflected by the ground to the side and then back upward due to

the proximity of nearby buildings or rising terrain, etc, to return through the top of the disc in a continuous circulating induced flow with increasing speed. The recirculation causes a loss of rotor thrust (lift) due to the decreased angle of attack with blade angle and rotor RPM being considered as constant. The helicopter requires an increase in power and collective to maintain the hover and avoid sinking to the ground. Hence, with recirculation present more power is required to hover compared to the power required when recirculation is not present.

Recirculation may be increased by hovering at lower heights where the down flow deflection is greater than when hovering at a greater height. Increased helicopter weight requires more power to hover and therefore creates a greater recirculation. Nearby buildings having a solid mass, have a greater influence than trees where the outflow can more readily escape. It may be that only part of the rotor disc is affected, which can cause a partial recirculation with attendant control problems.

*See* **Adverse pressure gradient; Power curves – helicopter; Settling with power; Vortex ring state.**

**Recompression shockwave** A recompression shockwave is formed when the transonic airflow over the aircraft is decreased to subsonic speed with aircraft deceleration. It is also known as an incipient shockwave.

*See* **Shockwaves.**

**Rectified air speed** *See* **Calibrated air speed.**

**Rectilinear flight** Rectilinear flight is an alternate name for straight and level un-accelerated flight at plus one 'g'.

**Reduction factor** *See* **Index units.**

**Re-entry speeds** Speeds greater than the hypersonic upper limit of Mach 25.0 are considered to be re-entry speeds associated with the Space Shuttle and other atmospheric re-entry vehicles.

**Reference area** Reference area is a term commonly used by aerospace engineers to describe wing area or frontal area in their calculations. It is termed  $S_{REF}$ . In the lift or drag formula familiar to pilots it is simply the letter 'S', the wing reference area.

*See Wing area.*

**Reference axes** The three axes around which the aircraft manoeuvres and transits are known as the reference axes; these are the OX longitudinal axis, the OY lateral axis, and the OZ normal axis.

*See Axes.*

**Reference Datum** The reference datum is the location on the aircraft from which all stations are measured for calculating the aircraft's centre of gravity. It is also known as the station zero.

*See Datum.*

**Reference speeds** For extracting the required performance from the aircraft, pilots use many reference speeds. Following are a few of the more common terms in use:

- $V_X$  – maximum angle of climb speed
- $V_Y$  – maximum rate of climb speed
- $V_{MC}$  – minimum control speed (for engine-out flying) indicated by a blue line on the ASI
- $V_{NO}$  – maximum normal operating speed (top of the green arc on the ASI)
- $V_{NE}$  – never-exceed speed (the red line speed on the ASI).

*See V-speeds.*

**Reference wing area** The complete area of the wing projected out from the fuselage centreline to the wing tips is considered to be the reference wing area.

It is also known as the gross wing area.

**Reflex camber airfoil** To provide a download force on the aft section of a wing, the wing is shaped with the mean camber line curving upwards towards the trailing edge; this is reflex camber. An advantage is the centre of pressure remains almost stationary reducing the moment about the aerodynamic centre, which reduces any unstable nose down pitching moments, common on conventional airfoils. Reflex camber can be an advantage on tailless aircraft.

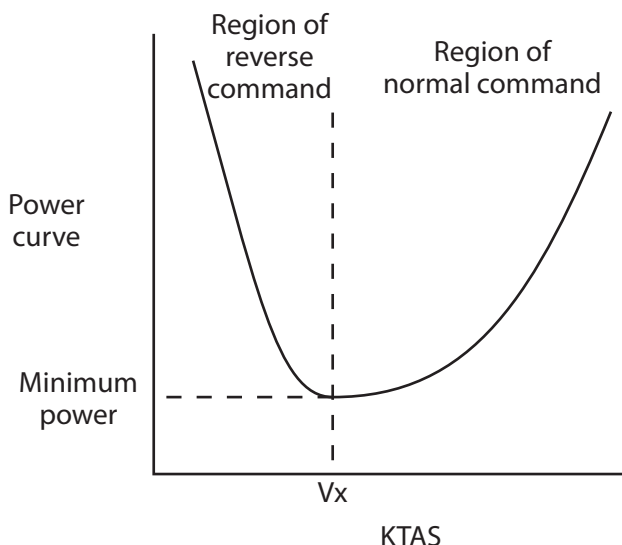
Reflex camber is also known as negative camber.

**Refracted boom** An aircraft flying at supersonic speed produces a Mach cone, which on reaching the ground is heard as a sonic boom. However, the upper part of the Mach cone moves skyward and normally dissipates. In the presence of high cosmic winds at approximately 150 000 feet altitude, the Mach cone can be refracted back towards Earth reaching ahead of the aircraft and is then heard as a distant rumble, if at all.

See **Sonic boom; Overpressure; Blunt nose re-entry vehicle.**

**Region of normal and reversed command** In the region of normal command, the aircraft will increase or decrease speed with a corresponding increase or decrease of power. The aircraft is said to be speed stable. **Diagram 70, Region of Normal & Reversed Command**, shows the region of normal command to the right of  $V_x$  minimum power.

The region of reversed command is to the left of the  $V_x$  minimum power. It is frequently termed as being on the backside of the power curve. As the aircraft's speed decreases from cruise speed to below the minimum power speed ( $V_x$ ) more power is required to maintain level flight due to the increase of induced drag with increasing angle of attack.



**Diagram 70, Region of Normal & Reversed Command**

**Region of reversed flow** At high angles of attack usually into the stall region, the airflow above the wing breaks away in turbulent flow. The higher-pressure air flowing under the wing may flow around the trailing edge and then forward on to the aft, upper surface of the wing. This is the region of reversed flow. It is also known as the dead air region.

*See* **Recirculation; Reverse flow region.**

**Relative airflow (RAF)** The relative airflow is the resultant direction of the airflow relative to the aircraft's flight path, the airflow over the propeller, and helicopter rotor blades. The effects of upwash and downwash over the wings are generally ignored and the average direction of the airflow is considered to be in the opposite direction to which the aircraft is traveling. Lift is found to be at right angles to the relative airflow and drag is parallel to it.

The free-stream airflow velocity is unaffected by the passage of the aircraft and therefore is not under the influence of the relative airflow. The relative airflow over a helicopter's rotor blades is affected by rotor RPM only – forward movement has no influence.

The relative airflow velocity is the same speed as the aircraft's forward speed but in the opposite direction; the vector is labeled on diagrams with three arrowheads to show its direction of application, as shown for example on propeller diagrams in this work.

It is also known as the free stream relative airflow, resultant airflow, or relative wind.

**Relative density** Relative density is significant when converting equivalent air speed to true air speed. It relates the density at altitude with density at sea level. The formula is as follows:

$$\text{Relative density } (\sigma) = \rho/\rho_0 = \frac{\text{density at given altitude}}{\text{density at sea level (ISA conditions)}}$$

Where  $\rho$  = density at altitude

$\rho_0$  = density at Sea level (standard density).

See **Density ratio; Standard density.**

**Relative pressure** The air flowing over the wing changes to the local pressure; either positive below the wing or negative above. The local pressure on the wing's surface compared to the free stream static pressure is known as the relative pressure, represented by the symbol  $P-P_0$ .

See **Pressure differential.**

**Relative wind** The relative wind is the velocity of the free-stream airflow measured some distance ahead of the aircraft. It is considered to be the same speed as the aircraft's true air speed in flight, but not necessarily aligned with the OX longitudinal axis.

**Relaxed static stability** Jet fighter aircraft with relaxed static stability are usually characterized with a smaller tail fin, a rear centre of gravity, and a longer nose cone. An automatic flight control system and its aerodynamic flight controls allow the aircraft's natural inbuilt, longitudinal static stability to be reduced for greater manoeuvrability; this is the relaxed static stability.

The Lockheed Martin F-16 Fighting Falcon is one example of an aircraft designed with relaxed static stability; due to its centre of lift located forward of the centre of gravity the plane is statically unstable in the pitch with deadbeat stability.



**Research aircraft** Following the end of World War II, America and to a lesser extent, England and Europe placed a great deal of emphasis on high-speed and high altitude flight. In the USA, NACA later NASA, the US Navy, Air Force, and aircraft manufacturing companies have been and still do, operate the most advanced experimental aircraft on research programs. The early programs were designed to study high-speed flight in the transonic, supersonic, and hypersonic speed range and to research effects of high altitude flight. The data gained by the American research paved the way for the space program and the development of today's modern aircraft.

Experimental research aircraft, used to verify wind tunnel data, are part of the aerodynamicist's tool kit along with modern high-power computers and wind tunnels. Some aircraft were custom built specifically for a given research project with only one or two examples built. Other aircraft were taken off the production line and customized in some way to flight-test specific ideas and to increase the ability to fly faster, higher or safer. Some aircraft were lost in accidents, others only made the few flights necessary to collect the required scientific data through their on-board computers and via telemetry signals to the ground stations.

The USAF in conjunction with NACA/NASA operated the X-series aircraft; the US Navy and manufacturing companies in association with NACA/NASA operated other research aircraft.

By no means complete, the following is a brief summary of some of the research aircraft used in America and England to investigate problems associated with aerodynamics:

## ***Bell X-1***

The Bell X-1 must be the most famous of the X-planes for being the first aircraft to exceed the speed of sound, Mach 1.0 with General Charles (Chuck) Yeager (1923–) at the controls on 14 October 1947. The flight took place at Edwards Air Force Base, California, the home of American test flying. The experimental research plane was propelled by a rocket engine after being released from its Boeing B-50 Superfortress mothership. The wind tunnel research at NACA Langley performed by Ezra Kotcher proved that short straight wings would be ideal to keep them out of the bow shockwave at supersonic speed; sweptback wings came later. The Bell X-1 and X-1A to D models were used to penetrate the speed of sound and investigate thermal problems and stress analysis for future high-speed jet fighter aircraft. A speed of Mach 2.5 was eventually reached with the Bell X-1 series aircraft.

NACA engineer John Stack was instrumental in the initial design stage for the Bell X-1.

See **Speed of sound (a)**.



**Photo:** The Bell X-1B research aircraft is a close relative to the Bell X-1. The X-1B sits under the port wing of the Valkyrie research/bomber in the National Museum of the US Air Force in Dayton, Ohio.

### ***Bell X-2 Starbuster***

The Bell X-2 Starbuster was built for continuing research into transonic and supersonic flight at speeds up to Mach 3.0, with an emphasis on aerodynamic heating at high-speed.

The first of the two aircraft to fly made its first flight on 18 November 1955. Between them, they reached a maximum altitude of 126 200 feet and a speed of Mach 3.196 (2094 MPH) in 1956. The Bell X-2 was the first aircraft to reach and exceed Mach 3.0, with USAF Captain Milburn Apt (1924–1956) at the controls on his one and only flight in the X-2. He lost control of the aircraft due to inertia coupling and died in the ensuing crash. Both aircraft were destroyed during research flights.

See **Aerodynamic heating**.

### ***Douglas X-3 Stiletto***

The Douglas Model 499D X-3 Stiletto were used to investigate turbo-jet engines, short-span (low aspect ratio) double-wedge wing and tail surfaces at very high altitudes, inertial coupling, stress research and the study of aerodynamic heating at speeds up to Mach 2.0. However, it was not as fast as intended. The X-3 was constructed of Titanium, as were the X-15 and other supersonic aircraft. Its first flight was on 15 October 1957 with twenty flights completed by NASA. The X-3 paved the way for the design of the North American F-100 Super Sabre and Lockheed F-104 Starfighter. In fact, data from the X-3's low aspect ratio, trapezoidal wing was employed in the Starfighter's wing design.

**See Aspect ratio; Aerodynamic heating.**



**Photo:** The Douglas X-3 was not as fast as its streamlined look indicated. Here, it is surrounded by other research aircraft in the National Museum of the US Air Force in Dayton, Ohio.

### ***Northrop X-4 Bantam***

Two Northrop X-4 Bantams were built, making approximately sixty research flights in total; the first flight was made on 15 December 1948 from Edwards Air Force Base. The X-4s were small experimental jet research aircraft built to study the low-speed characteristics of sweptback wing, semi-tailless aircraft operating at transonic speeds. They were similar in design to the German Messerschmitt Me163 and the British DeHavilland DH108 Swallow research aircraft.

See **Sweptback wings**.



**Photo:** The Northrop X-4 Bantam was a small swept-back research aircraft. Note the extended diverter air brakes. The Valkyrie stands over it in the NMUSAF in Dayton, Ohio.

## ***Bell X-5***

The Bell X-5 made its first flight on 20 June 1951, flown by Bell's chief test pilot Jean 'Skip' Ziegler (1920–1953). It was the first US aircraft to flight test the swing-wing concept, which was later used on the General Dynamics F-111 bomber, Grumman F-14 Tomcat and other swing-wing aircraft. The flight-testing confirmed the NASA wind tunnel research for reduced drag and improved performance due to wing sweep. The world's and Bell's X-5 first in-flight swing-wing movement was achieved on 16 July 1951, flown by Jean 'Skip' Ziegler.

A forerunner to the Bell X-5 program was the German Messerschmitt P.1101, designed for swing-wing flight-testing with ground adjustable wing sweep positions but it never achieved flight. United States troops discovered the project at the end of World War II at Oberammergau in Germany.

See **Swing-wing aircraft**; **Variable-geometry wing**.



**Photo:** The Bell X-5 was built to flight-test the swing-wing concept. NMUSAF Dayton, Ohio.

### ***Douglas D-558-1 Skystreak and D-558-2 Skyrocket***

The Douglas D-558-1 Skystreak and D-558-2 Skyrocket were developed with the Bell X-1 to provide some of the first in-flight data on transonic flight. Douglas test pilot John F. Martin (1907–1977) was the first pilot to fly the D-558-2 Skyrocket in 1948. However, test pilot Scott Crossfield (1921–2006) was the first person to achieve Mach 2.0 on 20 November 1953 after the D-558-2 Skyrocket was air launched from a Boeing B-52 mothership. The Skystreak was used to research air loads on the craft, which the early wind tunnels could not measure. Vortex generators were also tested on the D-558. The Skyrocket was a swept-wing jet-powered aircraft, which was later converted to rocket power. Edward Henry Heinemann (1908–1991) designed the Douglas Skyrocket.

The D-558 aircraft were not X-planes as such, they were operated jointly by the US Navy and NACA; the X-plane designation was reserved for US Air Force aircraft.

Unfortunately, Howard Clifton Lilly (1916–1948) became the first NACA test pilot to die when the D-558-1 Skystreak he was flying crashed on 33 May 1948.

**See Transonic airflow.**



**Photo:** The Douglas D-558-2 Skyrocket, the first plane to achieve Mach 2.0. Located in the Smithsonian National Air & Space Museum, Washington, DC.



## ***North American Aviation X-15***

The North American Aviation X-15 was the third type in the X-series specifically built for high-speed research after the Bell X-1 and X-2 series aircraft; the other X-planes were built for research into other flight problems.

The X-15 was a purpose-built, part spacecraft/research aircraft designed to explore the problems associated with re-entry from space at hypersonic speed. It provided flight data on the physiological aspects of high-speed/high-altitude flight and its effect on the aircraft's structure and flight controls and the effect of aerodynamic heating due to skin friction and shockwave formation.

Titanium was used in its construction to withstand the temperature rise of high Mach number flight. It was the first



**Photo:** The North American X-15 rocket plane located at the National Museum of the US Air Force in Dayton, Ohio, is the world's hypersonic speed record holder of Mach 6.80.



aircraft designed to fly above 50 nautical miles (100 km) altitude, considered the beginning of space. Three X-15 aircraft were built for research. One blew up on the ground during an engine test and was destroyed; the second was rebuilt as the X-15A-2 after suffering a landing accident in 1962.

Test pilots Albert Scott Crossfield (1921–2006) a US Navy pilot and later with North American Aviation, flew the X-15 on its first flight on 8 June 1959 and William Harvey Dana (1930–20?) made the last flight of the program on 24 October 1968.

The X-15 was the first aircraft capable of achieving hypersonic flight (Mach 5+) and to enter the edge of space. On 17 January 1963, NASA pilot Joe A. Walker (1921–1966) flew it to a record altitude of 271 000 feet (82600 meters) and became the first American pilot to receive astronaut wings. He later broke his own altitude record on 22 August 1963 when he climbed to 354 200 feet (67.08 miles). This altitude record still stands for a manned aircraft with conventional flight controls. The edge of space is defined by the French, Federation Aeronautique Internationale as an altitude of 328 084 feet (100 kilometers or 62.137 statute miles).

On 3 October 1967, USAF Major William K. 'Pete' Knight (1929–2004) USAF flew the X-15A-2 to a record hypersonic speed of Mach 6.80 (4520 MPH) on 23 October 1967. The record for the fastest speed achieved, excluding spacecraft.

### ***Northrop Grumman X-29A***

The Northrop Grumman X-29A-FSW was built for NASA to research the advanced technologies for future aircraft. It had wings of composite construction to test the concept of forward sweep. It was the first aircraft to fly with thin supercritical wings, a canard foreplane and digital fly-by-wire control system. Flight research was performed at the Dryden Flight Research Centre at Edwards Air Force Base, California in a joint program with NASA, the USAF and DARPA (Defence Advanced Research Projects Agency). It made its first flight in 1984 piloted by Charles Sewell and it was operated until 1992, when a total of 374 flights were made by the two examples built.



**Photo:** The Grumman X-29A flight-tested the forward swept wing concept. This one is on display at the National Museum of the US Air Force in Dayton, Ohio.

### **X-43A**

In more recent years NASA's unmanned X-43A research aircraft powered by a scramjet engine achieved a world speed record of Mach 9.6 (nearly 7000 MPH) on 16 November 2004. In a joint program between NASA's Dryden Flight Research Centre and Langley Research Centre the \$US230 million Hyper-X program culminated in the X-43 reaching Mach 9.6.

Three aircraft were built, each one performing only one flight each.

*See* **Hypersonic aerodynamics.**

## Convair XF-92A

Captain Charles (Chuck) Yeager first flew the Convair XF-92A on 18 September 1948. It was the first American delta wing aircraft to be built and was used to obtain data on delta wing aircraft operating in the transonic region. The result of this research was used for the design and development of the F-102, F-106, and B-58.

See **Delta wing**.



**Photo:** The Convair XF-92A was America's first delta wing aircraft. It is now on display at the National Museum of the US Air Force in Dayton, Ohio.

### *Ames/Dryden AD-1*

A research aircraft known as the AD-1 was designed jointly by NASA's Ames and Dryden Research Centres to test the concept of the oblique wing. The chief designer of the AD-1 was Robert T. Jones (1910–1999) at the Dryden facility; wind tunnel testing was carried out at the Ames facility with a first flight of the AD-1 on 21 December 1979. The AD-1 is a single-seat jet-powered aircraft with a rigid wing pivot-mounted on top of the fuselage. For low-speed flight, the wing sits at right angles to the fuselage in the conventional way. As speed increases one wing sweeps forward and the other moves aft in unison. A sweep angle of  $60^\circ$  is selected for flight at Mach 1.4 and  $70^\circ$  for flight at Mach 2.0. Dr. Richard Vogt (1894–1979) devised the AD-1's initial concept.

See **Oblique wing**.

## ***North American Aviation XB-70A Valkyrie***

The North American Aviation XB-70A Valkyrie was designed as a Mach 3.0 bomber for the US Air Force, but it never went into production. However, it did become the world's largest experimental research aircraft with two prototypes used for flight-testing.

As a research aircraft, it was used to study the characteristics of large supersonic aircraft, in particular, the study of stability and handling characteristics and associated aerodynamics. Reaching a speed of 2000 MPH and an altitude of 70000 feet, it was the first large aircraft to do so. It made its first flight on 21 September 1964 and was operated until 1969. One of the prototypes suffered a mid-air collision with a Lockheed F-104 Starfighter in a publicity photo formation flight, which downed the two



**Photo:** The Valkyrie was the world's largest research aircraft. It is now on display at the National Museum of the US Air Force in Dayton, Ohio.

aircraft. The Starfighter pilot was Joe Walker (1921–1966) who died in the accident.

The Valkyrie's configuration included a plain 65° delta wing with wingtip panels capable of drooping down to 65 degrees at high-speed to improve stability. The drooped panels effectively reduced the wing area, which had the advantage of reducing the centre of pressure's aft movement, thus preserving longitudinal stability. The drooped tips also boxed in the area between the tips, lower wing surface and the engine/weapons bay; this helped to produce a favourable shockwave in this region which induced additional lift known as compression lift (or wave riding) which provided a very good lift/drag ratio.

A canard stabilizer was also used to assist longitudinal stability. The canard provided the necessary up trim when flying at high Mach numbers and high angles of attack due to the rearward shift in the centre of lift, which caused a nose-down moment. The need for major trimming was eliminated.

*See* **Stability; Compression lift.**

### ***North American F-100 Super Sabre***

A production F-100 Super Sabre aircraft was used to research the instability problems of inertial coupling due to the type being the first aircraft to suffer this problem.



**Photo:** A Super Sabre was used to investigate the problems of inertial coupling. This aircraft is located in the National Museum of the US Air Force in Dayton, Ohio.



### ***Northrop/NASA M1-F1 and M2-F2***

The M2-F2, HL-10, and the X-24 were all lifting-body research aircraft designed for the study of re-entry problems of spacecraft. All three craft were devoid of any wings to speak of and obtained their lift, rather surprisingly, from the shape of their fuselage, which produced a high lift/drag ratio.



**Photo:** The Northrop M2-F2 lifting body in the Smithsonian National Air & Space Museum, Washington, D. C.

The Northrop/NASA M2-F2 was designed by NASA engineer Robert D. Reed (?–2005) and made its first flight after being released from a Boeing B-52 mother ship on 12 July 1966 at Edwards AFB, California. The craft had a flat top, bulging underside, raised cockpit and two fins. The M2-F2 was the powered version following the un-powered M1-F1. It was dubbed the ‘flying bathtub’ due to its appearance. Milton O. Thomson (1926–1993) was the first pilot to fly the Northrop/NASA M2-F2.

See **Lifting bodies**.

### ***Northrop HL-10***

The Northrop HL-10 lifting body was designed by NASA Langley as a follow up to the M2-F2 and made its first flight on 13 November 1968. The Northrop HL-10 was more streamlined than the M2-F2 with a cambered bottom, round top, three fins, and a flush cockpit in the nose.

The designation of 'HL' stood for horizontal landing.

See **Lifting bodies**.

### ***Martin-Mariette X-24***

The third research aircraft was the Martin-Mariette X-24 lifting body, originally known as the SV-SP. It made its first free flight on 17 April 1969 at Edwards AFB (previous flights were made under tow). The X-24 'A' and 'B' models followed the X-24.



**Photo:** The Martin-Mariette X-24B lifting body is now on display at the National Museum of the US Air Force in Dayton, Ohio.

### ***General Dynamics F-111***

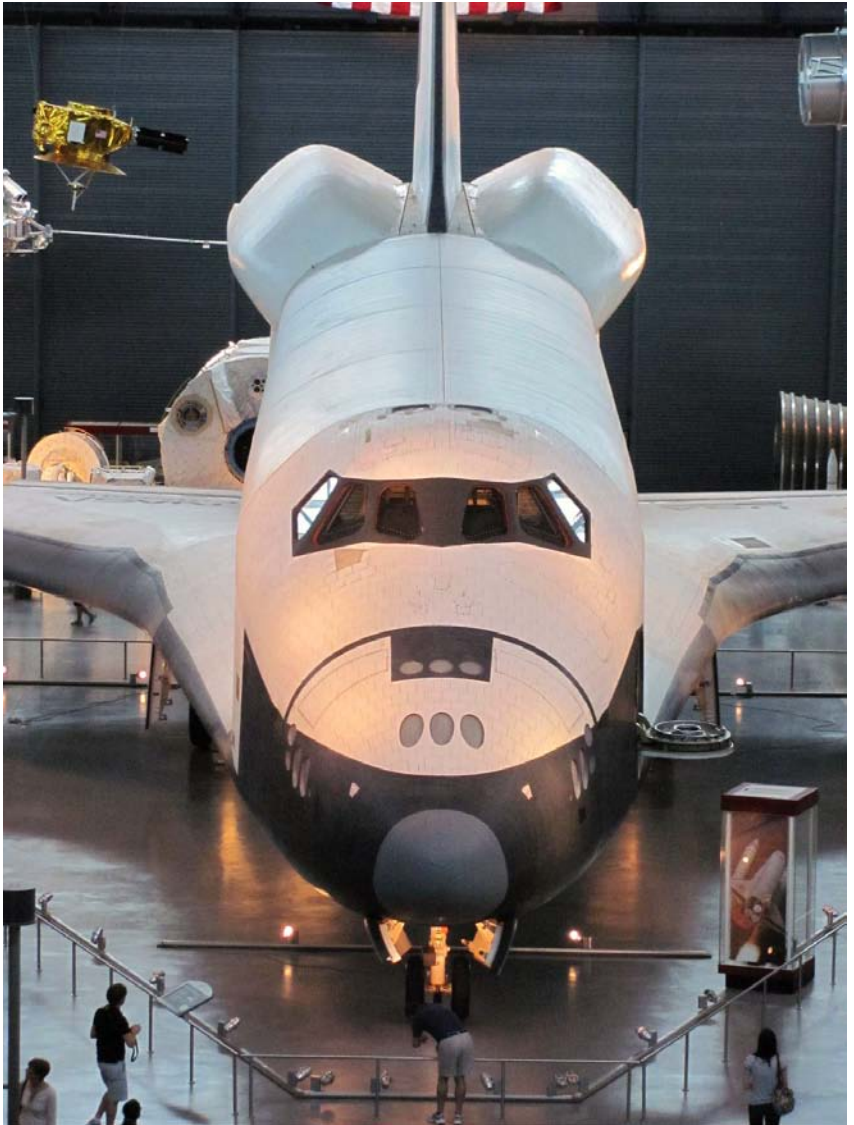
A production F-111 was converted to flight test the mission adaptive wings concept.

See **Mission adaptive wing**.

### ***Rockwell International Space Shuttle***

Rockwell International built the well-known Space Shuttle not for aerodynamic research but for space research. However, it does have some aerodynamically interesting features with its double delta wing with an inboard sweep angle of  $81^\circ$  reducing to  $45^\circ$  on the outer leading edge. It also has a blunt nose to control the bow shockwave on re-entry. Four elevons mounted on the trailing edge control the craft's pitch and roll during the atmospheric stage of flight. The rudder can be split to act as a diverter airbrake. The temperature rise on re-entry, up to  $1500^\circ\text{C}$ , requires the use of special heat-resistant tiles. The first flight made by the *Enterprise*, occurred on 12 August 1977 after being air launched from a specially modified Boeing 747. Its orbiting cruise speed is approximately Mach 27.0 or about 17300 MPH.

See **Blunt nose re-entry vehicle; Double-delta.**



**Photo:** The Rockwell International Space Shuttle Enterprise on display at the Smithsonian National Air & Space Museum in Washington, DC.

***Republic YF-84A***

A Republic VF-84A Thunderjet prototype jet fighter was employed to evaluate vortex generators.

See **Vortex generators**.

### ***North American Rockwell T-2 Buckeye***

On 24 November 1969, a North American Rockwell T-2 Buckeye flew for the first time with a Whitcomb/Langley supercritical wing for research purposes.

An LTV (Vought) F-8A Crusader SCW also flew with a modified supercritical wing (SCW) to flight test the new concept. The supercritical wing raised the aerodynamic efficiency by 15% over a conventional wing. It first flew as such from Edwards Air Force Base on 31 March 1973 for the Dryden Flight Research Centre.



**Photo:** A young girl inspects the Rockwell T-2 Buckeye onboard the USS Midway aircraft carrier, at the Midway Museum in San Diego harbour, Ca. A similar model T-2 Buckeye was used to flight-test supercritical wings.

See **Supercritical airfoil**.



### ***Rockwell/DASA X-31***

Rockwell International Corporation (USA) and MBB/DASA (now EADS) of Germany, to research supermanoeuvrability for fighter type aircraft, built the joint USA/German experimental Rockwell/DASA X-31 aircraft.

Using vectored thrust, the aircraft is capable of flying with angles of attack up to  $70^\circ$  and perform extremely tight turns and other combat manoeuvres beyond the capability of most fighter aircraft. It also demonstrated extremely short take-off and landing capabilities due to the vectored thrust. Approach angles of attack are double the normal  $12^\circ$  used by conventional fighter aircraft. Two aircraft were built with one being lost after the first flight on 11 October 1990. The aircraft layout is one with a low aspect ratio delta wing with a canard stabilizer.

**See Super-manoevrability.**

## ***Fairey F.D. 2***

The British built Fairey Delta F.D.2 research aircraft first flew on 15 February 1956 and proved the concept of tailless delta wing aircraft. The High Speed Flight at RAE Bedford operated it on supersonic research.



**Photo:** The Fairey Delta 2 (WG 777) on display in the RAF Cosford Museum, UK.

### ***BAC 221***

The BAC 221 research aircraft was a modified Fairey Delta F.D.2. It was in fact a mini version of the BAC Concorde complete with a drooping nose cone and its original wing was replaced by a miniature ogee-ogive planform chosen for the Concorde. It made its first flight on 1 May 1964 and was withdrawn from test flying in 1973. The Royal Aircraft Establishment (RAe) Bedford operated it.

Flown by Peter Twiss (1921-?) the aircraft claimed an official (FAI) World speed record of 1132 MPH (Mach 1.80 at 38000 feet) to become the World's first aircraft to exceed 1000 MPH, on 10 March 1956. However, it must be remembered the Americans exceeded this speed several years earlier with their research aircraft, but they did not make any claim to official speed records with the FAI.



**Photo:** The BAC 221 (WG 774) flight-tested the Concorde wing after its conversion from the speed record-braking Fairey F.D. 2. It is on view at the Fleet Air Arm Museum, Yeovilton, UK.

### ***Avro 707A, B & C***

The four examples built of the Avro 707 delta wing research aircraft, were scale models of the upcoming Avro Vulcan bomber (the World's first delta wing bomber). The Avro 707A made its first flight on 4 September 1949 and was used to investigate the flight characteristics and stability of delta wing aircraft flying at low-speeds.

The Avro 707C is a two-seat version, which arrived at the end of the Vulcan research program, so it was used instead to test an early form of fly-by-wire flight control system starting in 1960, and side-stick control, while stationed at the Royal Aircraft Establishment.



**Photo:** One of the two Avro 707C research aircraft on display in the RAF Cosford Museum, UK

### ***Boulton Paul P.111A***

The BP.111A was built for research into delta wing planforms, with a first flight in 1950, at Boscombe Down. It had the smallest airframe capable of flight with a delta wing, jet engine, and ejector seat. Initially built as the BP.111, it was modified to an 'A' model following a wheels-up landing.



**Photo:** The Boulton Paul BP.111A research aircraft retired at the Midland Air Museum, Coventry, UK.



### ***Bristol 188***

The Bristol 188 research aircraft was built of stainless steel in the late 1950s for research into airframe thermal soaking with temperatures up to 300°C and to gain high-speed data for the proposed Avro 730 Mach 3.0 supersonic aircraft. Due to its twin-engine's high fuel consumption, the flights were of insufficient endurance and combined with the inability of the aircraft to attain its design cruise speed of Mach 2.0, the high temperatures required for thermal soaking were not achieved. However, the research data that was gained was used to advantage in the design of the BAC Concorde and the development of its Rolls Royce Olympus 593 engines.

Two aircraft were built, the first being scrapped and this sole survivor is now located at the RAF Museum Cosford, UK.



**Photo:** The Bristol 188 on display in the RAF Cosford Museum, UK.

### ***DeHavilland DH 108 Swallow***

The DeHavilland DH 108 Swallow was designed in 1946 from a converted DH Vampire F1 fuselage to be a tailless, swept wing aircraft. It was given the unofficial name Swallow by the British Ministry of Supply. It was England's first transonic, swept wing aircraft.

Its original purpose was to be a testbed for the proposed tailless DeHavilland Comet airliner, which was on the drawing board at that time. However, the decision to build the Comet with a conventional tail left the DH 108 available for other research work and so it was used to investigate the handling characteristics of a swept wing aircraft at low-speeds up to the transonic region.

Three DH 108s were built with a first flight on 1 May 1950. Between them, they accumulated a total of 480 test flights. They successfully proved the concept of the swept wing tailless aircraft and went on to become the first British aircraft, along with test pilot John Derry, to exceed Mach 1.0 on 9 Sept 1946.

All three aircraft were lost in fatal accidents. The first accident occurred on 27 Sept 1946 when Geoffrey Roal DeHavilland Jr. (1910–1946) experienced unstable pitching oscillations in the second prototype during a high-speed run over the River Thames estuary. The second loss involved the third prototype on 15 February 1950, followed by the first prototype being lost during stall tests on 1 May 1950.

**See Tailless aircraft; Sweptback wings.**



### ***Short SB5***

The Short SB5 research aircraft was designed and built to research the low-speed handling characteristics of the sweptback wing for the up-coming English Electric P.1, the prototype for the E.E Lightning sweptback wing aircraft. The wing sweep angle could be changed (on the ground) from an initial 50 degrees to a final 69 degrees. It was also flight tested with a T-tail and conventional low-mounted tailplane. Notice the fixed undercarriage.



**Photo:** The Short SB5 on display in the RAF Cosford Museum, Cosford, UK.

### ***Handley Page HP.115***

The Handley Page HP.115 was built to test the concept of a slim delta wing operating at relatively low air speeds used for take-off and landing. The research results contributed to building Concorde.

Test pilot Jack Henderson took the HP.115 aloft on its first flight on 17 August 1961 from RAe Bedford, England. The craft, with 80-degree leading edge sweepback, proved the delta wing concept would fly extremely well at low-speed and high angles of attack without the need for leading edge or trailing edge devices. Vortex lift ensured the aircraft would fly safely and under control. The low aspect ratio delta wing produces considerable drag at low take-off and landing speed; it was very speed unstable meaning it needed more power to fly at ever decreasing speeds below a certain figure. Auto-throttles were installed on Concorde to counter any un-noticed speed loss on the approach and they are now found on just about all transport aircraft. The HP.115 had a fixed undercarriage because it was built for low-speed work only – it did not need to be retracted.



**Photo:** The HP 115 flight-tested the delta wing at low air speeds. The plane is now on display at the Fleet Air Arm Museum at Yeovilton, UK.

## ***Supermarine Spitfire***

The Royal Aircraft Establishment (Farnborough) to investigate the adverse effects of transonic flight used a Supermarine Spitfire PR Mark XI. Flown by test pilot Squadron Leader Tony Martindale, he conducted high-speed dives from 40000 feet, reaching Mach 0.92 on one flight, the highest speed recorded by a piston-engine aircraft. Mach tuck and elevator reversal became evident – the cause of many fighter aircraft being lost in combat. Recovery was accomplished when the Spitfire reached a lower altitude where the air was warmer and denser.



**Photo:** Spitfires were built in many versions, and a Mark PRXI was used to research compressibility problems. This Spitfire PRXI is displayed at the National Museum of the US Air Force in Dayton, Ohio.

### ***Junkers Ju 287-1***

The Junkers Ju 287-1 was built in Germany during the Second World War to investigate the forward swept wing concept. However, it never reached the flight testing stage.

**Resistance formula** The term resistance in aerodynamics refers to the drag of an aircraft moving through the air. The formula is written as below:

$$R = K \rho V^2 S$$

Where  $R$  = resistance in Newtons

$K$  = coefficient dependant on shape  
and found by experiment

$\frac{1}{2}$  = a constant

$\rho$  = air density in  $\text{kg/m}^3$

$V^2$  = velocity in meters per second

$S$  = frontal area in square meters.

*See Drag equation.*

**Resonance** Helicopters can suffer the effects of air or ground resonance where the helicopter is induced to rock violently from side to side resulting in an unstable roll over. Air resonance, known as pendulum mode, is less common but it can still lead to lateral rocking and loss of control.

*See Ground resonance.*

**Restoring couple** A restoring couple consists of two parallel forces of equal and opposite magnitude. One of the forces multiplied by the arm of the couple determines the moment.

**Restoring moment** A moment is a turning force. The horizontal tailplane acts as a restoring moment when the aircraft's nose pitches up or down. It depends on the tailplane force and its distance from the centre of gravity.

*See Moment.*

**Resultant centre of pressure** *See Centre of pressure.*

**Resultant force** Due to the aerodynamic reaction on the wing, the resultant force is tilted slightly rearwards relative to the direction of flight. The resultant force is the result of aerodynamic reaction produced by the wing moving through the air. It can be divided into two or more vectors emanating from the same point. For example on the aircraft's wing, the resultant can be divided into the vectors of lift and drag. A propeller or helicopter rotor blade has the resultant divided into the vectors of lift/drag and thrust/torque.

An inspection of *Diagram 67, Propeller Forces in Cruise Flight* and also *Diagram 72, Reverse Thrust Forces – Propeller*, shows the resultant force with its associated thrust/torque and lift/drag vectors.

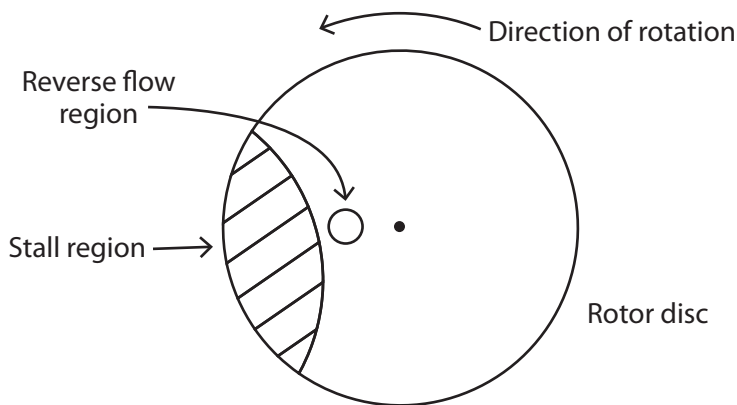
The resultant force is also known as the total reaction.

See *Diagram 67, Propeller Forces in Cruise Flight; Diagram 72, Reverse Thrust Forces – Propeller; Total reaction.*

**Retreating blade** When a helicopter's rotor blade moves towards the rear position on the rotor disc, it becomes the retreating blade. At the blade tip, its velocity is equal to the helicopter's forward speed minus the rotor RPM reducing to zero speed towards the blade root and becoming a negative value at the blade root itself. At the blade root, the airflow is in the reverse direction compared to that at the tips, that is, the airflows from the trailing edge to the leading edge. Lift and thrust is seriously affected.

**Retreating blade stall** A helicopter in forward flight has its retreating blade moving at a slower speed through the air than the advancing blade. Although the rotor RPM or angular velocity remains the same on both (or all) blades, the airflow velocity along the length of the blade varies with maximum speed at the rotor blade tips and least at the blade roots. The blade pitch can be considered as being constant along the blade's length.

However, because the rotor disc is tilted forwards during forward flight, the retreating blade has to move uphill to the rear of the rotor disc. The angle of attack at the blade tips becomes greater than at the roots. **Diagram 71, Retreating Blade Stall**, shows a counter-clockwise rotating rotor system. The shaded area shows the general pattern of the stalled rotor disc area. A smaller region of reverse flow occurs on the retreating blade's root area.



**Diagram 71, Retreating Blade Stall**

The resulting blade stall causes the helicopter to vibrate and it may pitch nose-up or oscillate in the pitching plane. If allowed to continue, the helicopter rolls over, usually to the left-hand side but it can be either way. Although retreating blade stall is a phenomena of relatively high forward speed and contributes to the limit on the never exceed speed ( $V_{NE}$ ), it can also manifest itself at high rotor



RPM at low forward speeds, high 'g' manoeuvres, turbulence or full and abrupt control movements. At  $V_{NE}$ , controls problems become more relevant than structural problems.

Nearly half of the retreating rotor blade span outer portion can be affected at high cruise speed. On reaching the maximum lift coefficient (stall angle), the blade drops down due to less lift being produced. The disc area affected is from about the 10 o'clock position where the stalling commences to about the 7 o'clock position where it ceases, viewed from above on an anti-clockwise rotating disc.

*Compare with **Advancing blade concept**.*

**Reverse flow region** Reverse flow is found on a helicopter's retreating blade root when the helicopter's forward speed reaches a certain air speed.

With increasing forward speed, the airflow over the retreating rotor blade root becomes the same as the helicopter's forward speed. The two opposing airflow-speeds cancel out and there is no airflow over the rotor blade root area. A further increase in helicopter speed and the airflow passes over the blade's trailing edge towards the leading edge – a reverse flow direction. The affected blade area increases from the blade root out towards the blade tips with increasing forward speed and a loss of rotor lift (thrust) is apparent on the part of the rotor blade with reverse flow. [Note here that the blade is un-stalled in the reverse flow region and the maximum lift coefficient is not exceeded].

To compensate for this loss of lift, the outboard section of the rotor blade must increase its share of the lift by increasing the angle of attack – the blade flaps down more with increasing angle of attack. Eventually an angle of attack is reached where the retreating blade stalls; the helicopter then pitches nose-up and rolls side to side. This phenomenon is known as retreating blade stall.

The reverse flow only occurs on the retreating blade root, which causes stress on the blade due to a pitch-up twisting moment. A

small amount of lift force develops on the upper surface and with an opposing down force on the lower surface; the turning moment causes the blade to twist placing stress on the rotor blade. To reduce the stress caused by the twisting, part of the trailing edge may be cut away to reduce the affected blade area.

The opposing airflow's speed reduces to zero speed at the outer edge of the reverse flow region. The airflow's velocity over the rotor blade increases from the zero speed at this location to a maximum speed at the blade tips. Rotor RPM could theoretically be increased to reduce the effect of reversed flow, however, the rotor RPM does have its limits on the helicopter's forward speed.

*See Diagram 71, Retreating Blade Stall, and text; Mu ( $\mu$ ) ratio & barrier.*

**Reverse pitch** In reverse pitch the propeller blades are placed at a negative angle of attack to provide reverse thrust to aid braking after landing. The first production aircraft to have a reversible pitch prop was the German heavy fighter, Dornier Do 335 Pheil, with a push/pull configuration (the front prop was reversible). The plane first flew in October 1943 and was designed by Dr. Claudius Dornier (1884–1969) following his patent for the push/pull configuration in 1937.

**Reverse pressure gradient** *See Adverse pressure gradient.*

**Reverse taper** The concept of reverse taper was not successful due to the wing's weight increasing outboard towards the wing tips, which presented structural problems. The XF-91 Thunderceptor is an experimental version built with reverse taper wings to test swept wing stall problems.



**Photo:** The Republic XF-91 Thunderceptor with reverse taper wings is on show at the National Museum of the US Air Force, Dayton, Ohio.

**Reverse thrust forces** Reverse thrust is achieved by turning the propeller blades about thirty degrees past the fine pitch stop to a negative angle of attack, to assist the wheel brakes for stopping the aircraft after landing.

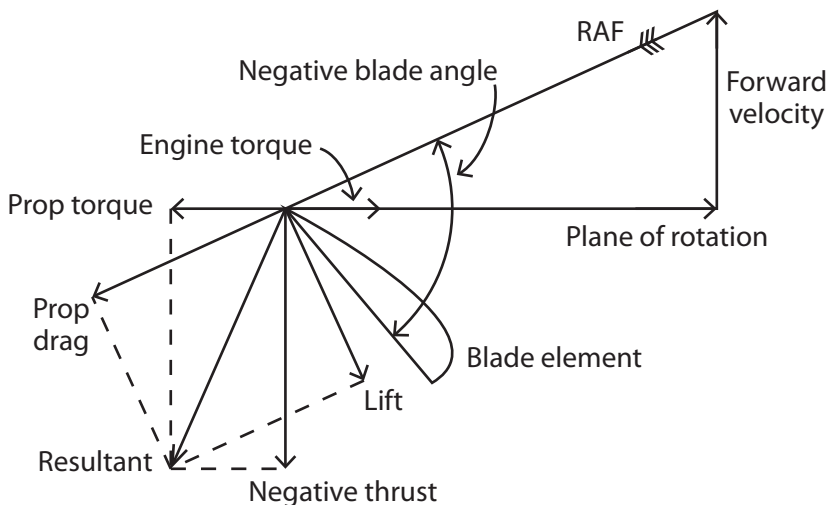
In the reverse thrust position, the resultant force on the propeller blades is acting in a rearward direction, the same as when the propeller is windmilling. During reverse thrust operations, propeller torque is equal and opposite to engine torque and acts

along the plane of rotation, as shown by **Diagram 72, Reverse Thrust Forces – Propeller.**

The rearward component of the resultant force produces negative thrust to provide a very effective air brake, assisted by the quite considerable propeller drag. With the aircraft decelerating during the landing roll, the forward velocity vector will reduce. This will cause the negative angle of attack to reduce also, resulting in a decrease of negative thrust. Hence the need to use reverse thrust early in the landing roll to take advantage of its greatest effect.

Because the propeller blades are at a negative angle of attack when operating in the reverse mode, the propeller will be less efficient than at normal forward motion pitch settings. However, full efficiency is not essential because reverse thrust is only applied for a few seconds during the landing roll. Reverse thrust may be referred to as braking pitch.

See **Braking pitch; Negative propeller thrust.**



**Diagram 72, Reverse Thrust Forces – Propeller**

**Reversed airflow** When a wing is in a stalled condition, the airflow from the increased pressure under the wing can flow around the trailing edge and forward over the upper surface, causing a loss of lift in the separated region on the rear half of the upper wing surface.

The reversed flow is also known as burble or the reverse pressure gradient.

**Reversed controls** An applied control deflection, usually in the rolling plane, on an aircraft flying at exceptionally high equivalent air speeds (EAS) causes the aircraft to roll in the opposite direction to that commanded. The wingtip twists due to a large aileron deflection distorting the wing. If the leading edge is warped downwards, the angle of attack is reduced and the aircraft rolls in the reverse direction. The wing is said to experience aeroelastic distortion.

*See* **Aeroelasticity; Flutter; Flexural aileron flutter; Forward sweep wings; Mass balance; Divergence.**

**Reversible process** A reversible process in a thermodynamic property is one that can be reversed by very small changes in the system without loss of energy. The property is in thermodynamic equilibrium due to the extremely small transitional change from the initial state to the final state of the process. It can also be stated following a reversible process, the process can be reversed without creating any change in the system or its surroundings.

Since it would take an infinite amount of time for the reversible process to finish, perfectly reversible processes are impossible. However, if the system undergoing the changes responds much faster than the applied change, the deviation from reversibility may be negligible. In a reversible cycle, the system and its surroundings will be exactly the same after each cycle.

The opposite process is an irreversible process where finite changes occur and the system is not in equilibrium.

*See* **Thermodynamics; Isobaric; Adiabatic process; Open cycle; Entropy; Isothermal.**

**Reynolds number** The Reynolds number (Re) is the dimensionless coefficient related to the ratio of inertia force to the kinematic viscosity force. It relates the importance of fluid viscosity effects, the shear stress (skin friction drag) of the air flowing over the surface of the wing and its association with scale effect of models in wind tunnels. It is also used in calculations associated with lift coefficient, and drag coefficient.

Reynolds number varies directly with air density, airflow velocities, length of the body (distance aft from wing leading edge) and it varies inversely with kinematic viscosity. A high Reynolds number is due to a high velocity, fluid flow over large wing chord area, low viscosity and at low altitude where the high inertia or dynamic forces are more dominant than viscous friction forces. For example, a high Re of 3 million ( $3.0 \times 10^6$ ) is a typical figure for a light aircraft flying with flaps up. Relatively low Re also applies to aerodynamics of small chord wings flying at relatively low velocity and at high values of kinematic viscosity, which are found at high altitude. Under these conditions, the low viscous or friction forces predominate, and at higher Re, the inertial forces dominate. Very low Re is not applicable to aerodynamics – only to biologists when studying microorganisms and blood cells.

Reynolds number reflects the characteristic of the surface boundary layer (laminar or turbulent) which determines the maximum lift coefficient and the transition point. **Diagram 73, Reynolds Number V. Lift Coefficient**, shows the Reynolds number increase with higher lift coefficients. The boundary layer airflow will be laminar at relatively low true air speed and Re 500 000 or less, as opposed to a turbulent flow of Re 500 000 or greater. However, the disadvantage of low Reynolds number is an adverse pressure gradient and laminar separation bubbles, which may occur causing increased form drag and reduced maximum lift coefficient.

The boundary layer flow will change to a turbulent flow at some higher value of Re, around 3 million transitioning to about 5 million as air speed increases due to the airflow gaining more

kinetic energy, which in turn produces a higher value of maximum lift coefficient. With increased Re, boundary layer separation is also delayed, again aiding the airflow to produce the higher maximum lift coefficient.

The maximum lift coefficient and Reynolds number both decrease with a gain in altitude due to a reduction in air density; higher altitude produces less kinetic energy because kinetic energy is determined by air density and true air speed, which varies with altitude. Laminar flow airfoils such as the NACA 4-digit airfoils are one of the results of wind tunnel research associated with Reynolds numbers.

Air density, true air speed, chord length and viscosity all have an effect on the final Reynolds number, which can be calculated thus:

$$\text{Reynolds number} = \frac{\text{Inertia force}}{\text{Viscous force}} = \frac{\rho V L}{\mu}$$

Where:  $\rho$  = air density

$V$  = air velocity (FPS)

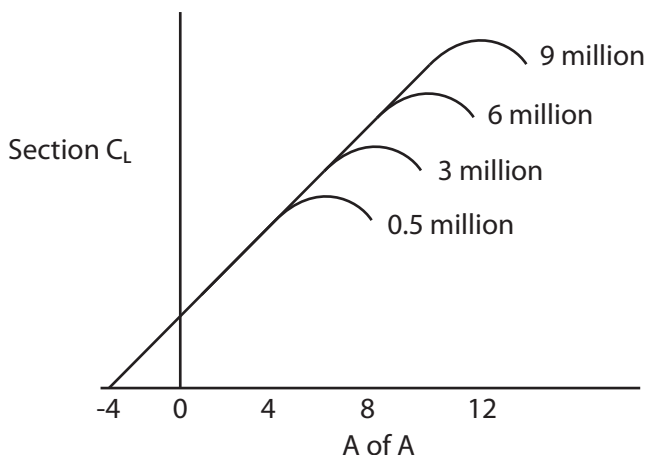
$L$  = chord length

$\mu$  = kinematic viscosity of fluid (sq. ft. per second).

The Reynolds number is of great importance to aerodynamicists in wind tunnel research work and aircraft design. Testing scaled-down models of aircraft in wind tunnels can result in the recording of erroneous data, where the models surface area and the velocity of the airflow must be taken into consideration. Therefore, an allowance must be incorporated in the test results to allow for this scale error; this is where the Reynolds number obtained in the tests is required to be as close as possible to the full size aircraft. The use of the models (with smaller wing chords) requires the wind tunnel to be operated at a higher atmospheric pressure or at an increase in wind flow velocity to compensate for this error. Max Munk (1890 – 1986) used a variable density wind tunnel at Langley Memorial Aeronautical Laboratory, which was in operation from 1923 for research into aeronautical data at high Reynolds numbers.

The Englishman, George Gabriel Stokes (1819–1903) first introduced the theory in 1851, which dealt with the flow of water through pipes. Osborne Reynolds (1842–1912) devised the Reynolds number system in 1883, twenty years before the Wright brothers' first flight. However, its importance became apparent in aerodynamics circa 1910, to compensate for scale error and other design calculations.

See **Scale effect; Viscosity.**



**Diagram 73, Reynolds Number V. Lift Coefficient**

**Reynolds, O.** Osborne Reynolds (1842–1912) was born in Belfast, Ireland. He studied mechanical engineering at Cambridge and Manchester Universities, graduating in 1867 as a physicist and engineer, specializing in fluid dynamics.

He is best remembered in aviation for his studies on the effects of fluid density in a streamlined flow, which he expressed as the Reynolds number system in 1883, twenty years before the Wright brothers' first flight. However, its importance in aerodynamics became apparent in about 1910, when Lord Rayleigh, John William



Strutt (1842–1919) introduced the Reynolds number as a non-dimensional factor in fluid flow calculations and to compensate for scale error.

Osborne Reynolds is also credited for discovering the fact, airflow changes from a laminar flow to a turbulent flow at the transition point, as the air flows over the wing's upper surface.

*See* **Reynolds number**.

**Reynolds stress** *See* **Shear stress**.

**Rhomboidal wing** *See* **Diamond wings**.

**Rhombus wing** A rhombus wing has a symmetrical double wedge wing section for use on very high-speed supersonic aircraft.

**Right-handed propeller** An aircraft propeller rotating clockwise as viewed from the rear of the propeller is termed a right-handed propeller.

*See* **Handed propellers**.

**Rigid rotor** Helicopters have rigid, semi rigid or fully articulating rotor heads. A rigid rotor system, being a hingeless rotor system has blades that cannot flap up and down, or lead and lag, but only feather or change pitch. A cyclic change in angle of attack takes place as the blades rotate around the hub. With no flapping motion, severe stress is placed on the rotor hub as the rotor blades attempt to flap up and down and drag. Stress is relieved by the use of rubber bearings. The advantage here is the rigid rotor system has the greatest control power of the three systems.



**Photo:** The Eurocopter/Bolkow BK 117 is a twin-turbo-engine helicopter with a four-blade rigid rotor system, which rotates anti-clockwise when viewed from above.

Lockheed engineer Irvin Culver developed one of the first rigid rotor systems, which was tested and used on a variety of helicopters in the 1960–70s.

The rigid rotor is also known as a hingeless rotor.

See **Rotor heads**.

**Rigidity & precession of gyroscopes** A gyroscope (or gyro for short) is a fast spinning rotor mounted in a series of gimbal rings such that it maintains a set direction in space. Gyros have two properties – rigidity and precession.

Rigidity, also known as gyroscopic inertia, describes the reluctance of a spinning gyro to a change in its direction of spin axis. Hence, it obeys Newton's first Law of Motion. The gyro wheel has its greatest mass located around the outer rim to provide greater rigidity. Angular velocity (RPM) is also an important characteristic of the gyro. The higher the RPM and the greater the mass, the greater will be the rigidity; it will be more resistant to external torque forces. Rigidity is therefore directly proportional to the gyro's RPM and its moment of inertia.

The second property of the gyro is precession. The gyro is designed to maintain a set direction relative to space as stated above. However, if an external torque force is applied, it will turn briefly in the direction of the force and then, turn about its axis at  $90^\circ$  to the torque force in the direction of rotor spin. The precession will remain constant for as long as the torque force is being applied. If you hold a spinning gyro by its axis and then try to tilt the axis laterally the gyro will turn  $90^\circ$  in azimuth as if the force was applied at the front or rear edge instead of at the top or bottom edge.

If the gyro is deflected from its pre-set direction, this is known as gyro wander, which can be classed as either apparent or real wander. Real wander can be caused by a physical deviation such as friction in the bearings, rotor imbalance, or some other mechanical interference. Apparent wander is caused either by the gyro being transported over the surface of the Earth (as in a plane in flight) or by the Earth's rotation; however the gyro maintains its fixed direction relative to space, not the Earth.

Wander in the horizontal plane is referred to as drift and in the vertical plane as topple. Free gyros with a horizontal axis can also drift. This too is caused by the Earth's rotation and the drift varies with the sine of the latitude.

It is in the study of flight instruments, especially the artificial horizon, direction indicator and the turn and bank or turn co-ordinator instruments that we require knowledge of the gyro and its properties of rigidity and precession. Gyros are also used in the inertial navigation systems of larger aircraft and autopilots.

An appreciation for the gyroscopic effects of propellers is also required pilot knowledge. A tail-wheel type aircraft during take-off will experience gyroscopic precession, as explained above and will tend to yaw and swing, usually to the left; it depends on the direction of propeller rotation. In addition, as the aircraft manoeuvres in flight, gyroscopic precession can cause the aircraft's nose to rise or drop depending on the direction of turn or pitch.

The use of gyros dates back to the 18<sup>th</sup> Century. The first gyro was designed by Serson, a British scientist in 1744 to stabilize ships' sextants when no horizon was visible. However, in 1852, Jean Bernard Foucault (1819–1868) a French physicist, is also credited with early experimental work on the gyroscope. Gyros were installed in torpedoes as early as 1896. In 1908, Anschultz of Germany further perfected the gyro. In aviation history, Elmer Sperry (1889–1930) is well known for introducing gyroscopic flight instruments having founded the Sperry Gyroscope Company in the USA in 1911.

**Rigging angle of incidence** The angle between the wing's chord line and the OX longitudinal axis is the riggers angle of incidence.

It is also known as the angle of inclination or angle of incidence.

**Ring wing** *See Flat Annular wing.*

**Rogallo, Dr. F. M.** Dr. Francis Melvin Rogallo (1912–2009) in conjunction with his wife Gertrude invented and patented the now famous Rogallo wing used by many hang glider pilots. Dr. Rogallo was employed by NACA/NASA from 1936 to 1970 as an aeronautical engineer. He was one of the first to earn an engineering degree at Stanford University in 1935. For his invention of the Rogallo/flexible wing, he was dubbed the Father of Hang Gliding.

*See Flexible wing.*

**Roll axis** This is an alternate term for the OX longitudinal axis about which, the aircraft rolls.

*See Longitudinal axis.*

**Roll coupling** An alternate name for inertia coupling.

*See Inertia coupling.*

**Roll stability** *See Spiral mode stability.*

**Roll subsidence mode** An aperiodic motion in the rolling plane, which is well damped. It is related to the Dutch roll and the spiral mode in the subset of lateral dynamics.

**Rolling damping moment** During a roll upset, the lift, drag and angle of attack all vary when the dropping wing meets the relative airflow at an increasing angle of attack, which will increase the lift on that wing. Conversely, the rising wing meets the relative airflow at a smaller angle of attack producing less lift. The difference in lift force between the two wings induces a restoring moment to level the wings back to equilibrium. The rolling moment is considered positive when the aircraft rolls right wing down in association with the angle of sideslip, which is also positive when the relative wind is to the right of the aircraft's centreline.

The rolling moment coefficient ( $L$ ) associated with the rolling damping moment has the units of force times length and is shown by:

$$L = C_i q S b$$

$$C_i = \frac{L}{q S b}$$

Where  $L$  = rolling moment ft.lbs

$C_i$  = rolling moment coefficient

$q$  = dynamic pressure, psf

$S$  = wing area, sq ft

$b$  = wing span, ft.

*See Dihedral angle; Lateral dynamic stability.*

**Rolling instability** Rolling instability is an upset around the aircraft's OX longitudinal axis.

Following a lateral upset, if the aircraft rolls a few degrees (around 5 degrees or so) lateral leveling will be restored by the inherent dihedral action and/or pendulum stability. Beyond a given limit, the aircraft may sideslip, yaw, and roll off into a spiral dive resulting in rolling or lateral instability.

*See Dihedral action/effect; Lateral dynamic stability.*

**Rolling plane** The rolling plane is the OX longitudinal axis around which the aircraft rotates when it rolls (turns). The roll is considered positive when the aircraft rolls to the right and negative when it rolls to the left.

**Rolling tailplane** A rolling tailplane performs the combined duties of pitch control around the OY lateral axis and roll around the OX longitudinal axis. A rolling tailplane is used on the North American X-15 research aircraft and the General Dynamics F-111 fighter/bomber, for example.

It is also known as a taileron.

**Rooftop** A lift coefficient curve with a gently sloping top at the maximum lift coefficient is known as a rooftop curve. A rooftop lift coefficient curve represents an aircraft with docile stalling characteristics. Likewise, the pressure pattern over the chordwise top surface of a supercritical wing is closer to an ellipse and is so described as a rooftop pressure pattern.

*Compare with* **Peaky**. *See* **Supercritical airfoil**.

**Rooftop section** A supercritical wing has a relatively flat upper surface, earning the name rooftop section. It is the opposite of a cambered airfoil found on conventional wings.

*See* **Supercritical airfoil**.

**Root** The root (in full – the wing root) is that part of the wing where it joins onto the fuselage and includes a small, unspecified portion of the wing itself. The wing root may be streamlined with a root fairing or on a supersonic aircraft, the wing may be blended with the fuselage.

**Rotational airflow** Rotational airflow ( $V_R$ ) is due to the propeller or rotor blade rotation. The rotational airflow varies along the length of the propeller or rotor blade, the vector A-B on *Diagram 67, Propeller Forces in Cruise Flight*, is shorter at the prop hub and longer at the blade tips. The vector A-B will also vary with RPM, being a shorter vector at low RPM and longer at higher RPM. For clarity, the vector is drawn to show the average RPM for the prop/rotor blade at a chosen blade chord.

**Rotational velocity** Rotational velocity is the speed of propeller rotation in revs per minute (RPM). This is shown on *Diagram 67, Propeller Forces in Cruise Flight*, as the vector A-B.

It is also known as the tangential velocity.

*See* **Vorticity**.

## **Rotor aerodynamics** *See Helicopter aerodynamics.*

**Rotor blade compressibility** When the main rotor tips of a helicopter's advancing blade reaches the critical Mach number, problems of compressibility arise, which cause an increase in rotor drag.

Rotor blade compressibility can be avoided by:

- limiting helicopter forward speed and therefore the advancing blade tip speed
- reducing helicopter forward speed in turbulence to avoid an unwanted increase in angle of attack in turbulent gusts
- limiting rotor RPM if possible to reduce the advancing blade tip speed
- operate at lower helicopter all up weight (AUW) in order to reduce thrust required and therefore enable lower angles of attack
- Operate at lower density altitude where less rotor thrust is required.

Thick, cambered airfoils with high lift coefficients are normally a disadvantage on helicopters because they decrease the critical Mach number. This is a reason for using symmetrical airfoils with lower lift coefficients, which helps to increase the critical Mach number and reduce rotor blade compressibility.

**Rotor blades** A helicopter's rotor blades provide the aerodynamic lift in the form of thrust to support the machine in flight.

The rotor blades mounted on most helicopters are of a symmetrical shape. A characteristic advantage of the symmetrical airfoil shape is the blade's twisting moments are reduced; therefore, the shallow spars can be built of lightweight materials using less internal bracing. This has the following advantage:



- changes in the blade's angle of attack results in little movement of the centre of pressure
- the aerodynamic centre and the centre of pressure are located on the blade's feathering axis and
- The centre of gravity is located close to the feathering axis.

The disadvantage of the symmetrical airfoil shape is:

- the maximum lift coefficient is reduced
- A large critical angle of attack is not possible due to the relatively sharp leading edge.

See **Feathering axis; Directional static stability.**

**Rotorcraft** The term rotorcraft includes all aircraft deriving their lift from rotating rotor blades, such as helicopters, gyrocopters and autogyros.

See **Rotary wing aircraft.**

**Rotor disc** The rotor disc is the total circular area swept by the rotor blades of a helicopter or autogyro. The disc area reduces with increased coning angle. With a decrease in rotor blade RPM the centrifugal force of the blades is reduced and the lift force raises the blade to a higher position – an increased coning angle. With the rotor blades arcing upwards, the blade tips will move slightly closer to the rotor hub, thus reducing the rotor disc area. With rotor disc area (S) being part of the lift formula, it follows that lift will be reduced with increased coning angle.

**Rotor disc attitude** The angle between the rotor disc and the horizontal reference line defines the rotor disc attitude.

**Rotor disc incidence** The rotor disc incidence is the angle between the rotor disc and relative airflow. This is the helicopter's equivalent to the angle of attack between the airplane wing's chord line and the relative airflow. The vectors A-C and A-D on **Diagram 68, Propeller Pitch**, show it.

**Rotor disc RPM** A helicopter's rotor disc RPM is required to be maintained within certain limits during flight. The limits are maintained by:

- the pilot's adjustment of power via the throttle
- increasing the RPM by using low blade pitch angles and reduced by the use of high pitch angles. The difference in rotor blade drag affects the RPM
- increasing the RPM will reduce the lift/drag ratio and vice versa. The location of the total reaction leaning ahead of or behind the axis of rotation during an autorotation
- inflow air entering from below the rotor disc produces greater drag, which reduces RPM
- The RPM also varies with changes in transient and static droop.

**Rotor drag** The aerodynamic drag due to the rotor blade rotation is known as rotor drag. **Diagram 67, Propeller Forces in Cruise Flight**, shows the rotor blade's aerodynamic drag acts in line with, but downstream of the relative airflow (vector A-C). The drag is at right angles to the lift force.

The diagram also shows the rotor torque operating in line with, but downstream of the plane of rotation (vector A-B) and at right angles to the total rotor thrust (TRT). At a constant rotor speed, the rotor blade torque is equal and opposite of the engine torque. The aerodynamic rotor drag should not be confused with the rotor torque, they are two different forces!

See **Aerodynamic drag; Diagram 67, Propeller Forces in Cruise Flight**.

**Rotor droop** Rotor droop or (low rotor RPM) is a phenomenon associated to the operation of helicopters.

During powered descents, rotor droop can be induced by the blades' root stalling; the rotor blades tend to cone upwards, reducing the rotor blade disc area due to insufficient centrifugal force to keep them in their normal plane of rotation. This leads to less rotor thrust and the RPM drops below a given limit; the engine will be incapable of driving the rotor blades to a sufficient speed to maintain flight. Rotor droop can induce a disastrously rapid descent to a bad landing!

**Rotor force** The upward force of a helicopter rotor disc is known as the rotor force. The rotor force is an alternate name for the resultant force, which is divided into the vectors of total rotor thrust and rotor torque and also lift and drag.

See *Diagram 35, Rotor Forces in Cruise Flight*.

**Rotor heads** Atop the mast of a helicopter there will be one of three types of rotor heads to support the rotor blades. These are the fully articulated system, the rigid rotor system, and the semi-rigid rotor system as listed below:

- a fully articulated rotor system consists of three hinges to allow full and free movement of the rotor blades.
- a rigid rotor system has blades that cannot flap up and down but only feather or change pitch.
- a semi-rigid rotor system has no flapping or drag hinges. However, some blade flapping and dragging is allowed due to the use of a flexible material in the rotor blade roots.

See **Rigid rotor; Semi-rigid rotor; Fully articulated rotor**.

**Rotor thrust** All rotary wing craft, helicopters and gyrocopters, etc, produce the required rotor thrust (or lift) for flight by accelerating a mass of air through the rotor disc. In normal flight, accelerating the airflow downward through the disc produces the thrust on a helicopter rotor; a gyrocopter accelerates the airflow upward through the disc. Greater mass per unit time produces greater rotor thrust. Rotor thrust is a vector of the total reaction force produced by the rotor blade. The combined rotor thrust from both or all blades is known as the total rotor thrust.

The two most accepted theories on how rotor discs or propellers produce thrust (or lift) is described in the sections on axial momentum theory and the blade element theory.

See **Total rotor thrust**; *Diagram 88, Total Rotor Thrust*.

**Rotor thrust/torque ratio** The rotor thrust/torque ratio is the helicopter's equivalent to an airfoil's lift/drag ratio. The best efficiency of the rotor blades are found when operating at an angle of attack which produces the most rotor thrust for the least torque, with the pilot's collective control down at its lowest position. It follows, the best lift/drag ratio will also be achieved when the best rotor thrust/torque ratio is being produced.

The term rotor thrust applies to a single rotor blade, as opposed to total rotor thrust, which applies to all the blades on the rotor disc.

See *Diagram 88, Total Rotor Thrust; Diagram 41, Hovering Forces*.

**Rotor tip advance ratio** See  $\mu$  ( $\mu$ ) ratio & barrier.

**Rotor torque** Newton's third law of equal and opposite reaction is well demonstrated in the helicopter's rotor torque. Through the transmission system, the engine imparts a turning or torque force to drive the rotor blades. The equal and opposite reaction to engine torque is for the rotor blades to try to turn the helicopter around in the opposite direction by an equal amount – that is rotor torque (or rotor drag). To balance these two opposing torque forces requires the use of the anti-torque tail rotor. Increasing or decreasing the force on the tail rotor can be used to turn the helicopter in the yawing plane.

The rotor torque is akin to the propeller torque.

See *Diagram 88, Total Rotor Thrust*.

**Rotary wing aircraft** A rotary wing aircraft is one, which derives its lift from rotating blades, as opposed to an airplane's fixed wings. The term rotary wing includes helicopters, gyrocopters, gyrogliders and autogiros, etc.

They are known also as rotorcraft.

**Royal Aircraft Establishment, Farnborough** The Royal Aircraft Establishment is located at Farnborough Airport, Hampshire, England. It is the birthplace of British aviation and the seat of aeronautical research. One hundred acres of Farnborough Common saw the first buildings erected from 1905 onwards, some now demolished with the passing of time.

The American born Colonel Samuel Franklin Cody (1867–1913) was involved with kite and glider flying, airships and ballooning with the HM Balloon Factory, which operated here from 1905 to 1908. He died nearby in a flying accident in 7 August 1913. Samuel Cody is believed to have made the first powered flight in Great Britain from Farnborough on 16 October 1908.

In 1912 the Balloon Factory was renamed the Royal Aircraft factory and in the same year, the Royal Flying Corps was founded here in April, the forerunner of the Royal Air Force.

The Royal Aircraft Factory, to avoid confusion with the same initials as the Royal Air Force, received a further name change in 1918 to the Royal Aircraft Establishment, a name that was to be around for many years until 1988 when it was updated to the Royal Aerospace Establishment (RAe).

Scientific flight-testing, research, and development of all things aeronautical were performed at Farnborough by the various departments in residence there. The Instrument Landing System (ILS) was developed at Farnborough along with research into radio and instruments, meteorology, armament and of course, aerodynamics to name just a few of the activities. An eight-foot wind tunnel (the largest in Europe for Mach numbers up to Mach 2.8) was used to research over 100 wings for Concorde.

Wind tunnels were an important part of the establishment, the first of which was built in 1916. Others that followed included a 24 feet wind tunnel, which opened on 4 April 1935 to be the largest wind tunnel in England and the largest return tunnel in Europe. A transonic wind tunnel was also used along with a high-speed variable density wind tunnel that opened in 1947. British researchers were not so interested in speeds above 600 MPH in the early days. However, Germany carried out in depth research into problems of high-speed flight and became world leaders. Dr. D. Küchemann (1911–1976), Dr. Adolph Büsemann (1901–1986) and several other German scientists were brought from Germany at the end of WWII to work at Farnborough when the importance of high-speed research became obvious. Several captured German WWII aircraft were taken to Farnborough for flight evaluation during and after WWII.

The RAe also used a second airfield at Bedford from 1946 onwards. It was known as RAF Thurleigh during WWII, home to the 306<sup>th</sup> Bombardment Group.

Famous aircraft such as the Bristol Brabazon, BAe/McDonnell Douglas Harrier were flight tested at Farnborough. The BAC 221, Avro 707, Handley Page HP.115 and the Fairey F.D.2 were based here for research programs into slender delta wings.

Farnborough is now home to the Air Accident Investigation Branch and the British National Space Centre. The wreckage from the DeHavilland Comet disasters of the 1950s was inspected here along with other aircraft accidents. Farnborough has now played an important part in Britain's aviation history for over one hundred years.

**Rudder** The rudder, located on the trailing edge of the vertical fin is used to manoeuvre the aircraft left or right around the OZ vertical axis, to maintain balanced flight and co-ordinate turns.

**Ruddervator** A V-shaped tail, or butterfly tail has two control surfaces that work in unison as elevators or in opposition as rudders, which are known as ruddervators.

See **Butterfly tail**.

# S

**'S' (wing area)** The area of the wing is commonly abbreviated with the letter 'S', as found for example in the lift formula:  $Lift = C_L \frac{1}{2} \rho V^2 S$ . The wing area may be described as gross or net wing area. The gross wing area is taken to be the whole planform area ignoring the placement of the fuselage or engine nacelles with the wings meeting at the fuselage centre line. The net wing area excludes the fuselage and engine nacelles.

**Safety margin** The difference in pitching moments between a full power climb and descending at idle power will cause the neutral point to vary its position by 7–8% MAC. Because of this variation in position, albeit by only a small amount, the aft centre of gravity location is always separated from the neutral point by the safety margin, which is normally considered a fixed distance, as opposed to the static margin, which is variable.

It is also known as the minimum static margin.

See *Diagram 12, Centre of Gravity Range; Diagram 78, Static Stability V. Angle of Attack.*

**Sails** The term sails is an alternate, but less common name for winglets.

**Sailplane** Sailplane is the correct name for a glider.

**Saw tooth** A saw tooth is located on the leading edges of a swept wing aircraft, usually on the main wing, however, some aircraft have a saw tooth on the tailplane as found on the McDonnell Douglas F-15 Eagle. The saw tooth is a break in the continuity of the leading edge where the outboard edge protrudes further forward than the inboard edge. Its purpose is to induce a strong vortex to form at the location and prevent the boundary layer airflow from



moving outboard towards the wingtips, which in turn, reduces the development of a strong ram's horn vortex.

A saw tooth is also known as a dogtooth, notch or rebated leading edge.

*See* **Rams horn vortex.**

**Scalar quantities** Scalar quantities have magnitude but no direction.

**Scale effect** Scale effect is the characteristic change in Reynolds number of a model under wind tunnel testing when compared to the full-size aircraft. To account for the difference in size between the model and the aircraft, a scale effect allowance is factored in. The use of variable density wind tunnel to test the model – with increased air density – helps to reduce the errors of scale effect.

The maximum lift and drag are the two most important aerodynamic factors, which are affected by scale effect and the change in Reynolds number.

*See* **Reynolds number; Diagram 73, Reynolds Number V. Lift Coefficient.**

**Schlieren photography** Schlieren photography allows the aerodynamicist to view shockwaves on models in the wind tunnel.

The German word Schlieren translates roughly to striations or streaking. The principle is based on the original work of the French physicist, Jean Bernard Foucault (1819–1868) in 1859 and later improved on by Professor Ernst Mach (1838–1916) in 1887, who was the first person to use it for studying shockwaves on bullets. Professor Ernst Mach improved the principle with the introduction of the interferometric technique; the Mach-Zehnder interferometer is still in use today.

Shockwaves, Prandtl-Meyer expansion waves and supersonic flow become visible on the screen as variations in the light and dark patterns. This is due to the changes in air density in the flow

– dark bands represent shockwaves and lighter areas represent the high-pressure side of the shockwaves.

**Scimitar propellers** Sweptback propellers work in the same manner as sweptback wings on a jet aircraft by increasing the blade's critical Mach number, allowing the prop to run at speeds closer to Mach 1.0, with reduced noise. Curved or scimitar shaped blades are an alternative to sweptback tips, which have a similar effect.

The difference in speed between the scimitar blade's root and tip area affect a different amount of air mass through the propeller in a given time. For this reason, the scimitar blades are designed with extra chord width around the 50% station (half way along the blade's length). At low-speed during take-off, more of the thrust is produced in this area of the blade. As the aircraft's speed increases, the major part of the thrust is produced further outboard along the curved span of the scimitar blade providing better efficiency at higher speeds where the effects of compressibility are reduced.

The scimitar shape is ideal for Propfans and they are becoming more popular on new generation turboprop aircraft such as the Aerospatiale ATR 72 and several new light aircraft models. The scimitar shape provides better thrust and aerodynamic performance on today's modern aircraft, but the shape is not a new idea. An inspection of photographs of early aircraft from around the time of World War 1 will reveal many propellers had a scimitar shape. Was it for aerodynamic or aesthetic reasons the early propeller designers chose the scimitar shape? After the WW1 era, the style faded away and propeller blades became straighter; but now once again we are seeing the scimitar shape return to our aircraft.



**Photo:** Scimitar-shaped propellers were common on World War I era aircraft but then went out of favour. They are now re-appearing again on modern aircraft. This photo shows the Lockheed C-130J Hercules cargo plane with six-blade scimitar propellers.

**Scimitar wing** A scimitar wing is a sweptback wing with a geometric planform that curves spanwise. The angle of sweep reduces from the root to the tip.

The scimitar wing is also known as a crescent wing.

**Scissor wing** A wing mounted on an aircraft that is able to swivel from the ninety-degree athwart ship position for take-off and landing, to a sweptback position on one side of the aircraft and a sweep forward position on the other side. The NASA/Ames AD-1 was used to flight test the scissor wing concept at NASA Dryden during 1979–82 and at Patuxent River for a proposed scissor wing supersonic airliner. Only one AD-1 aircraft was built.

See **Oblique wing**.

**Scramjet** A scramjet is a supersonic combustion ramjet engine.

In a normal ramjet, a shockwave forms in the engine's inlet at high supersonic speeds causing pressure losses; therefore, the ramjet is very inefficient below Mach 5.0. However, the use of a supersonic combustion (sc) chamber – making it a scramjet, boosts the engine's performance to speeds well beyond Mach 5.0. A scramjet is an air-breathing type of engine, burning ambient oxygen plus hydrogen-fuel in the compressor. Although the scramjet engine is highly efficient above Mach 5.0, it needs solid rockets to boost the aircraft to the scramjet's initial operating speed.

NASA's unmanned, scramjet-powered X-42 research aircraft claimed a World speed record of Mach 9.6 on 16 November 2004.

**Sealed control gap** A flexible strip seals the open gap between the control surface and the trailing edge of the wing to prevent drag inducing airflow from flowing through. In contrast, some aircraft benefit from a slot between the wing trailing edge and the leading edge of the control surface, making it a slotted surface.

See **Slotted aileron**.

**Sears-Haack body** A Sears-Haack body produces the least shockwave drag due to its varying cross-sectional area distribution, its blunted shape, and its fineness ratio, which is designed to suit the craft's cruise Mach number. They are used on aircraft, bullets, missiles, or rocket nose cones.

The rounded nose cone shape is designed from a mathematical formula; it is not just a geometric shape. All Sears-Haack nose cones are blunt bodies, not sharp points. The given length, volume and diameter of the nose cone determines the shape for minimum wave drag. The fineness ratio (or ratio of the nose length to its diameter) affects the amount of wave drag acting on the aircraft's nose at transonic or supersonic speeds. At fineness ratios beyond 5:1 drag increases due to increased skin friction and wetted area, counteracting the reducing wave drag due to fineness ratio. For all aircraft and missiles designed to operate in the supersonic speed range (or higher) require their nose cones to be designed in accordance with the Sears-Haack formula to limit the increasing nose pressure drag, a major factor above Mach 0.8.

The Sears-Haack body is the brainchild of two independent researchers, William Rees Sears (1913–2002) and Wolfgang Siegfried Haack (1902–1994).

It is also known as a Von Karman ogive after Professor Theodore von Karman (1881–1963).

*See* **Sears, William Rees; Haack, Wolfgang Siegfried.**

**Sears, William Rees** William Rees Sears (1913–2002) was an engineering student under Professor Theodore von Karman (1881–1963). He worked on airfoil theory investigating the changes in the shape and airflow velocity over airfoils, which became known as the Karman-Sears's theory.

Apart from other work on airfoil theories, he was also involved in the design of the Northrop P61 Black Widow night fighter and experimental flying wing aircraft. He is also noted for his theory in association with Siegfried Haack (1902–1994) known as the Sears-Haack theory, during the 1950s.

*See* **Sears-Haack body; Haack, Wolfgang Siegfried.**

**Secondary flight controls** The secondary flight controls consist of the flaps, spoilers, airbrakes, droops and slats, etc, for changing the lift, drag, or speed of the aircraft.

**Secondary flow** The airflow over the wings from the leading edge to the trailing edge is the primary flow. The secondary flow is induced to flow very closely around the wingtips due to the pressure difference above and below the wing. This induces a wing tip vortex, which produces induced drag. The use of drooped or raised wingtips pushes the secondary flow further outboard where it has less power to form a vortex flow. The wingspan is effectively increased artificially without changing its geometric dimensions. The increased effective span increases the wing's aspect ratio and thus, less induced drag is formed due to a weaker vortex.

*See* **Wingtip vortices; Induced drag.**

**Secondary structures** The aircraft's secondary structures are all items that would not be a hazard to continued flight if they failed, e.g., undercarriage doors, wingtip fairings, etc.

**Second Law of Thermodynamics** The 2<sup>nd</sup> Law of Thermodynamics deals with changes in temperature. The law introduces the existence of a quantity known as entropy and its tendency to increase over a period. This is another way to describe the loss of energy. Any system will never be 100% efficient at converting energy, as some heat energy is lost to its surroundings.

The physicist Rudolph Clausius described the law as determining the direction of energy flow from a hot source to a cold source and never in the reverse direction without producing some other effect. Applying this law to a jet engine for example, a limit is set on the maximum efficiency of the engine; a certain amount of heat energy will always be lost in the process of converting heat to power. The perfect engine with 100% efficiency is impossible to build. The thermodynamic and propulsive efficiencies are the two characteristics that define the overall efficiency of all jet engines.

See **Thermodynamics; Isentropic flow; Adiabatic process; Isobaric; Open cycle; Entropy; First Law of Thermodynamics.**

**Section angle of attack ( $\alpha_o$ )** The section angle of attack denoted by the Greek symbol of  $\alpha_o$  is the angle between the average relative wind and the wing's chord line. Due to the induced downwash the wing 'sees' the air flow as coming from a new direction, represented by the average relative wind vector on **Diagram 43, Induced Drag.**

The section angle of attack  $\alpha_o$  plus the induced angle of attack  $\alpha_i$  are the two parts of the angle of attack commonly referred to in textbooks.

See **Angle of attack ( $\alpha$ ); Diagram 43, Induced Drag.**

**Section drag coefficient ( $C_d$ )** The section drag coefficient replaces the more familiar drag coefficient when related to the angle of attack of a two-dimensional wing section. The 'C' chord length in the section drag coefficient formula replaces 'S' wing area. A lower case 'd' is used in place of a capital 'D' for section drag coefficient.

The section drag formula is shown as follows:

$$\text{Section } C_d = \frac{\text{Drag}}{\frac{1}{2} \rho V^2 C}$$

Where  $C_d$  = drag coefficient

$\frac{1}{2}$  = a constant

$\rho$  = air density kg/m<sup>3</sup> or lbs/cu.ft.

$V^2$  = aircraft speed in m/sec<sup>2</sup> or FPS<sup>2</sup>

$C$  = chord length in meters or feet.

*See Drag coefficient.*

**Section lift coefficient ( $C_l$ )** The section lift coefficient is the lift divided by the local chord and shows the relationship for a particular two-dimensional cross-section, or chord of the wing. It is assumed the wing is of an infinite span with a constant cross-section and the section lift coefficient cannot be defined in terms of total lift and wing area. Therefore, it must be defined as lift per unit span. This is shown by:

$$\text{Section } C_l = \frac{\text{Lift}}{\frac{1}{2} \rho V^2 C}$$

Where  $C_l$  = section lift coefficient

$\frac{1}{2}$  = a constant

$\rho$  = air density kg/m<sup>3</sup> or lbs/cu.ft.

$V^2$  = aircraft speed in m/sec<sup>2</sup> or FPS<sup>2</sup>

$C$  = chord length in meters or feet.

Note, 'S' wing area in the lift formula is replaced with 'C' chord length in the section lift coefficient formula.



An airfoil's section lift coefficient is represented by the symbol  $C_l$  (lower case l) as opposed to a full, three-dimensional wing with a finite span where the more familiar  $C_L$  symbol has a capital L.

A two-dimensional wing – more commonly known as an airfoil section – is one of infinite span, where the effects of the wingtip are considered irrelevant. A three-dimensional wing's lift coefficient is always less than a two-dimensional wing due to wingtip losses accounting for this difference. A constant chord wing's lift coefficient will be equal to  $0.93 \times C_l$  maximum and for a tapered wing, the lift coefficient is equal to  $0.95 \times C_l$  maximum.

Aerodynamicists may prefer a two-dimensional wing section for research purposes, as opposed to pilots who prefer three-dimensional wings!

*See* **Lift coefficient; Span loading parameter.**

**Self-induced stall** A self-induced stall occurs when the wingtips on a sweptback wing stall before the wing root and results in a nose pitch-up or a roll, or both.

Wingtip stalling is an inherent problem with some delta or swept-back wing configurations. At high angles of attack, separation of the airflow over the wingtip causes a loss in lift, extra drag, buffeting and adverse stability. The wing stalls, progressing from the tip inboard to the wing root – an undesirable situation – resulting in a further nose-up pitching motion taking the aircraft deeper into the stall or, if only one wingtip stalls, the aircraft will roll off into a spiral dive.

*See* **Stall.**

**Semi-rigid rotor** A helicopter's semi-rigid rotor system has the rotor blades attached to the rotor hub but without any flapping or drag hinges. However, some blade flapping and dragging is allowed due to the use of a flexible material in the rotor blade roots and the use of a hanging, teetering rotor hub. When the advancing blade moves forward, its angle of attack is reduced, with an increase in angle of attack on the retreating blade. Thus, dissymmetry of lift is alleviated.



**Photo:** Semi-rigid rotor systems have the least control power of the three rotor systems.

The Bell Jet Ranger has a two-blade semi-rigid, teetering rotor system. Unlike other rotor systems, the Jet ranger's rotor has no coning hinges but relies on the blade's flexibility to cone upwards under load.

**Semi-span** The semi-span is half of a full wingspan measured usually, from a wingtip to the aircraft's centreline. In some instances, the distance is measured from the wingtip to the side of the fuselage when calculating wing bending moments.

**Semi-stalled** A wing is semi-stalled when the airflow over the wing is still partially attached to the surface while the remainder of the flow has separated.

*See Stall break.*

**Sensible atmosphere** The top of the Earth's sensible atmosphere is considered to be at an altitude of 400 000 ft or 122 km; beyond this altitude, it is considered to be outer space. Within the sensible atmosphere all aerospace vehicles (Space Shuttle and missiles, etc) experience air resistance or drag, an important factor to be considered for re-entry vehicles.

However, at an altitude of 100 000 feet, the sky turns from a blue color to black and above this altitude it is considered to be the fringe of space.

*See Atmosphere; International Standard Atmosphere; Structure & composition of the atmosphere.*

**Separation bubble** A separation bubble may form on the wing's upper surface within the boundary layer.

The bubble may be a small percentage of the chord or a long bubble, where it can cover the whole length of the chord; its length is dependant on the Reynolds number and the pressure gradient. The short bubble has less effect than a large bubble; its effect is negligible on the pressure distribution. With increasing angle of attack, the bubble moves forward and can trip the laminar flow causing flow breakaway and a trailing edge stall. A second scenario is the leading edge causing separation at high angles of attack resulting in a leading edge stall when the flow fails to re-attach to the wing's surface.

A long bubble covering a greater percentage of the chord has more effect on the pressure distribution, in particular at the leading edge where the peak suction is reduced, decreasing the energy of the flow over the wing. The long bubble may form at moderate angles of attack and spread chordwise with increasing nose pitch-

up. The lift coefficient decreases resulting in a progressive stall condition. Long bubble separation and stall is more common on thin airfoils.

It is also known as a laminar separation bubble.

See **Laminar boundary layer**.

**Separation point & flow** The flow is considered to be separated when it is no longer attached to the upper surface of the airfoil and departs the wing at a tangent, considerably reducing the required lift, and increasing the unwanted drag.

The location in the boundary layer where the flow's velocity becomes stationary is known as the separation point. The separation point and region is located approximately mid-chord on the wing's upper surface at the junction between the laminar and turbulent boundary layers with the separation point in the centre of the region. This point is rather hard to define with accuracy due to the gradual change in the separation region. Like the centre of pressure, the separation point also moves around with changing angles of attack. With increasing nose-up pitch or leading edge icing the separation point moves forwards towards the leading edge. The separated flow increases drag, reduces lift, and causes burble and vibration as the turbulent flow (or wake turbulence) strikes the tailplane unit. A normal shockwave also induces flow separation on the wing's upper surface. The separation point and region is closely related to the transition point and region respectively.

The leading edge of a subsonic wing has a large radius curvature, which delays the flow separation and stall until a high angle of attack is reached at around  $15^\circ$ , or higher, this causes a trailing edge stall. However, most high-speed wings have a sharper, small radius leading edge, which induces a flow breakaway at the leading edge and hence a leading edge stall. The characteristics of a wing's flow separation depends on the wing's angle of attack, thickness,

leading edge radius, sweepback angle and speed, which also affects the location of the associated transition and stagnation points.

Professor Ludwig Prandtl (1880–1953) and his co-workers at the Gottingen Aeronautical Research Centre in Germany, researched boundary layer flow separation as early as 1904. Further work by Eastman N. Jacobs (1902–1987) at NACA Langley proved that the airflow separation over the wing occurred further aft on laminar flow airfoils, caused not by the actual airfoil shape but it was due to the pressure distribution about the airfoil.

***Diagram 89, Transition Point & Region***, shows the location of the laminar and turbulent boundary layer and the separation point where the laminar flow changes to a turbulent flow.

See **Stagnation point/line; Vortex lift**.

**Separation vortex** A separation vortex forms at the leading edge of a delta wing at high angles of attack and flows over the upper surface re-energizing the airflow to maintain the lift.

See **Form thrust**.

**Service ceiling** The service ceiling as defined by the American FAA, is the altitude where the aircraft's rate of climb has reduced to 100 feet per minute.

Compare with **Absolute aerodynamic ceiling; Combat ceiling**.

**Sesquiplane** A Sesquiplane is a biplane with two different wing spans. The most noticeable feature is the upper wing's overhang and short length of the lower wing, at about half the span of the upper wing.

The term sesqui (pronounced seskwi) is the prefix for one and a half.



**Photo:** This photo of a Fiat CR42 Falco clearly shows the short wing span of the lower wings. Located at the RAF Museum in Hendon, London.

**Settling** Settling occurs when a helicopter lifts-off the ground with insufficient power to support the machine out of ground effect. To overcome settling, the pilot must increase the rotor thrust with collective, to prevent the helicopter returning to the ground.

**Settling with power** A helicopter may settle with power when descending close to the ground when approaching to land.

The helicopter settles in its own downwash with a steep angle of descent and a rate of descent at approximately 300 feet per minute, while using more than 20% of the thrust horsepower available. The air speed will be less than the effective translational lift air speed. Raising the collective to arrest the rate of descent only aggravates the problem by increasing the rate of descent further. The recovery technique is to fly to the side away from the downwash airflow.

It is also known as power settling.

See **Vortex ring state; Recirculation.**

**Shaft axis** The shaft axis is the axis of a helicopter's rotor blade drive shaft, which remains fixed relative to the helicopter's rotor mast. The axis of rotation (of the rotor blades) is moveable within limits and is only in line with the shaft axis when the plane of rotation is perpendicular to the shaft axis.

**Sharp edged gust** A sharp edged gust is defined as the instantaneous vertical velocity found in a steady wind, measured in feet per second. The vertical component is considered to be instantaneous when the aircraft flies into the up or downdraught resulting in a sudden change in the aircraft's angle of attack. If the change is too sudden it could overstress the aircraft and cause structural damage.

Aircraft certified today under the relatively new FAA Part 23 rules allows penetration of gusts up to 50 feet per second at the maximum cruise speed ( $V_C$ ). To be able to withstand such gusts, the aircraft are built slightly stronger than the aircraft built under the old Part 23 rules, where the gust was considered instantaneous (sharp edged). However, under the new rule the vertical gust is considered to have a transition region where the full velocity of the gust builds up relatively gradually.

Aircraft are also designed to withstand acceleration up to the ultimate load factor (50% higher than the limit load factor) for a period of up to three seconds. Three seconds may not sound like a very long time to adjust to the gust, but a gust alleviation factor is also allowed for. When the aircraft enters a vertical gust, it does so relatively gradually, traveling several chord lengths and pitching up before the wing produces maximum lift. Both of these factors reduce any instant stress force on the wings. Therefore, the full intensity of the gust is a relatively gradual onset and not instantaneous. For a given vertical gust, the acceleration in 'g' experienced by the aircraft increases with aircraft speed. In addition, the load on the wings builds up gradually from the leading edge spreading aft towards the trailing edge. This is different to the abrupt increase of load caused by an elevator pitch-up. The degree of turbulence depends on the vertical gusts, which are felt as bumps during flight.

Hence, the turbulence penetration speed ( $V_B$ ) is the maximum speed allowed for flying through gusty, turbulent conditions; the manoeuvring speed ( $V_A$ ) is the maximum allowable speed at which an abrupt and full control surface deflection can be made without overstressing the aircraft.

*See Diagram 59, Manoeuvre Envelope.*

**Shear stress** The resistance in a viscous, boundary layer airflow close to a surface is measured by its shear stress. It is the cause of skin friction, and is also known as Reynolds stress.

**Shear viscosity** *See Viscosity.*

**Shock angle** The shock angle is the angle formed between an oblique shockwave and the flow's original direction of motion.

*See Diagram 15, Compression Flow & Oblique Shockwave.*



**Shock body** A shock body is mounted on the wing's trailing edge on some high-speed aircraft. It is a streamlined bulge designed to enhance the distribution of area rule.

Also known as speed bumps, Whitcomb body, or Küchemann carrots.

**Shock-boundary layer interaction** A shockwave forming on the wing's upper surface adversely affects the boundary layer and is the cause of transonic drag.

*See* **Transonic drag hump**.

**Shock cloud** A localized cloud caused by sudden changes in flow conditions in close proximity to a supersonic aircraft, notably in Prandtl-Meyer expansion fan for example, over the wings and canopy. Visible condensation appears within the expansion region forward of the shockwave; however, as the air flows through the shockwave the air is rapidly heated, which instantly evaporates the condensation behind the shockwave.

*See* **Prandtl-Meyer expansion fan**.

**Shock collars or eggs** Shock collars or eggs are alternate names for Prandtl-Glauert condensation clouds.

**Shock compression** The air flowing over an aircraft at supersonic speed on passing through a shockwave undergoes an instantaneous shock compression.

**Shock detachment distance ( $\delta$ )** The shock detachment distance is the distance between the nose of the body (aircraft) and the detached shockwave itself. The detached shockwave is a normal shockwave directly in front of the aircraft's nose, curved rearwards to become a bow shockwave, and weakens with distance from a point directly in front of the nose. It eventually evolves into

a Mach cone. The shock detachment distance is symbolized by the Greek lower case letter delta  $\delta$ .

*See* **Blunt nose re-entry vehicle; Detached shockwave.**

**Shock diamonds** Shock diamonds are highly visible bright segments in the hot, supersonic exhaust stream from afterburning jet engines or rocket engines.

Compared to ambient air pressure, which is relatively high at lower altitudes, over-pressure induces an over-expanded, compressed exhaust stream. Due to lower pressure at high altitude, the exhaust flow is under-pressure, which induces an expanded flow and the shock diamonds dissipate.

Initially, a normal shockwave is formed in the exhaust nozzle and the first shock diamond is formed there, caused by the increase in temperature igniting excess fuel in the exhaust stream. Small oblique shockwaves are formed continuously and reflect from the edges of the exhaust stream due to the alternating compressing or expanding airflow. Several shock diamonds are formed in succession but cease to exist when the outer edge of the exhaust stream mixes with ambient air.

Shock diamonds are also known as Mach diamonds or discs, thrust diamonds, or dancing diamonds.

*See* **Contact discontinuity.**

**Shock diffuser** Jet-engine aircraft flying at supersonic speed require specially designed jet-engine inlets with a diffuser. The inflow velocity at the engine intake must be reduced from supersonic speed to subsonic or low-speed supersonic flow with minimum energy loss through induced shockwaves to avoid inflow air temperature rise. A pointed, bullet-shaped diffuser is located in the jet engine's intake to control the shockwave formation; the shockwaves formed will be a choice of a multiple oblique shockwaves, or a single normal shockwave, depending on the design cruise speed of the aircraft and inlet performance requirements.

**Shock layer** An oblique shockwave is attached to and emanates from the nose of an aircraft flying at supersonic speed, and other protruding edges. With increasing Mach number, the shock angle becomes smaller and the shockwave itself converges closer to the aircraft's surface. At very high Mach numbers, the shock layer is that part of the supersonic flow between the oblique shockwave and the surface boundary layer.

With the Mach number increasing, there is a corresponding increase in temperature of the airflow. At Mach 5.0 and higher (hypersonic speeds), the temperature has risen to the point where the chemical reaction of the gas molecules and the viscous flow affects are very evident. At subsonic speeds the airflow above the boundary layer is considered to be an inviscid flow; however, with a supersonic airflow it becomes fully viscous producing increasing amounts of friction drag and hence, considerable heat with increasing Mach number. Therefore, the airflow in the shock layer is a source of heat, which is transferred to the aircraft's body.

*See Hypersonics.*

**Shock stall** A shock stall occurs at low angles of attack and at a high-speed, as opposed to an aerodynamic stall, which occurs at high angles of attack and low-speed.

An aircraft flying near its critical Mach number may produce a shockwave on the upper surface of the wing. The shockwave causes a sudden rise in pressure, temperature, and density and a decrease in the airflow velocity. The boundary layer thickens and experiences a high drag rise and a drop in the lift coefficient and hence, a loss of lift due to a change in pressure distribution and movement of the centre of pressure and pitching moment. Airflow separation causes turbulence (burbling) from the wing surface aft of the shockwave to strike the tailplane, imitating a high angle of attack stall. The lift coefficient curve rises to a peak where the shockwave forms and then decreases to a level near the subsonic value, as shown on *Diagram 51, Lift Coefficient & Mach Number*.

*See Buffet boundary; Diagram 51, Lift Coefficient & Mach Number.*

**Shockwave drag** An aircraft accelerating through the critical Mach number, about Mach 0.8, experiences a drag rise, which increases considerably to a value about three times the subsonic value. This is known as shockwave drag.

Shockwave drag has two components consisting of wave drag and boundary layer drag. The shockwave retards the airflow, increasing the pressure behind it and a thickening of the turbulent boundary layer with separation. Both have the effect of increasing the drag due to the shockwave.

See **Wave drag**.

**Shockwaves** During subsonic flight, pressure waves – like sound waves – are projected ahead of the aircraft ‘warning’ the air of the approaching aircraft. The air stream simply divides as it flows over and under the wings and around the aircraft. The airflow over the entire aircraft is subsonic with only small variations in air density. The centre of pressure is well forward on the wing, normally between 20–30% MAC. The rules of subsonic aerodynamics apply!

At a cruise true air speed of about Mach 0.8 – the start of the transonic speed range – the airflow speed-increases over the wing’s upper surface to reach Mach 1.0 or supersonic speed in the area of lowest air pressure at the point of maximum camber at approximately 50% MAC; this is known as the critical Mach number.

The airflow velocity at the critical Mach number is greater than the freestream airflow surrounding the aircraft. Compressibility effects will start to become evident and control of a subsonic aircraft may become difficult in the form of porpoising and Mach tuck in pitch.

At this speed, an incipient shockwave (or recompression shockwave) forms on the wing’s upper surface causing air behind it to experience a reduction in speed to remain subsonic, with a sudden rise in static air pressure, density and temperature as rapid compression takes place, as shown by *Diagram 74A*. Compression

behind the shockwave increases boundary layer separation, due to insufficient kinetic energy in the adverse pressure gradient. The centre of pressure moves aft, which can induce Mach tuck on subsonic airfoils, which have stronger shockwaves and a lower critical mach number than supercritical airfoils. The lift coefficient rises to its peak value; however, the drag – the bane of transonic flight – is increasing.

The incipient shockwave becomes more intensified as it evolves into a normal shockwave and starts to lean back at a decreasing angle and move rearwards as cruise speed increases. The airflow-speed in front of a normal shockwave is supersonic with a subsonic flow behind. The reduction in airflow-speed behind the shockwave results in less lift and a decrease in lift coefficient for the whole wing. Shown by **Diagram 74B**.

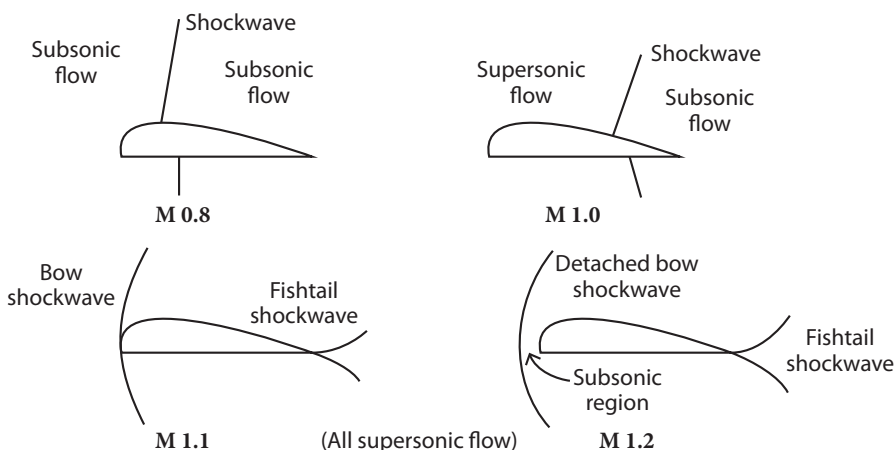
Exceeding Mach 1.0, the two original shockwaves on the upper and lower surfaces move aft to settle at the trailing edge as an oblique or fishtail shockwave. At an air speed of around Mach 1.2 – the high end of the transonic speed range – the wing will be totally immersed in a supersonic flow. Shown by **Diagram 74C**.

As the aircraft accelerates through the transonic region, parasite and wave drag reduce to a level similar to, but slightly greater than that found at subsonic speeds. At these speeds, wave drag becomes more prominent with parasite drag becoming less significant. The drag coefficient reduces close to its subsonic value and remains more or less constant at all speeds above Mach 1.3; this is clearly shown on **Diagram 21, Drag Divergence Curve**. The centre of pressure will also have moved rearwards with increasing Mach number causing a nose-down trim movement. Concorde counteracts this nose-down movement with fuel transfer to an aft holding tank to balance the aircraft.

The transonic region is a speed to be avoided for cruising due to the increased drag and fuel consumption. All aircraft must either cruise at subsonic speeds with Mach 0.9 being about the maximum limit or increase speed to cruise beyond Mach 1.4, or thereabouts.

Flying at a speed of about Mach 1.02, causes an increase in the pressure differential at the nose of the aircraft where the air becomes compressed and the temperature increases. For this reason, an aircraft flying at low supersonic speeds will have a relatively sharp nose, as found on Concorde and some jet fighters. The shock wave attached to the aircraft's nose, will initially be vertical, but with increasing flight speed the wave moves to an acute angled cone. It is relatively weak and the airflow between the aircraft's surface and the shock wave can reach high temperatures, which is transferred to the aircraft's structure, but is of less concern at low supersonic speeds compared to high supersonic speeds.

On re-entry vehicles, the Space Shuttle for example flying at hypersonic speeds, the temperature rise becomes unacceptable. For this reason, they employ a blunt nose, which causes the bow shock wave to move ahead of the nose at the Mach detachment speed ( $M_{DET}$ ). The detached bow shock wave, which is stronger than an attached wave, initially stands at a less acute angle before the angle decreases with increasing speed. The high temperature airflow between the bow shock and aircraft's nose has greater space to allow the heat to dissipate without heating the body excessively.



**Diagram 74, Shockwaves**

The high air temperature increases the speed of sound between the bow wave and the nose, and the pressure waves push the detached bow wave further forward.

Conversely, increased temperature may have the opposite effect. Behind the bow shock wave, the density increases with increasing temperature, and according to the conservation of mass, the volume of air behind the shock decreases, thus reducing the stand off distance between the bow shock and body, known as the small shock stand off distance. The aircraft's speed and nose profile and hence, its temperature determines if the bow wave detaches or not. Therefore, it can be seen the bow wave standoff distance varies with the local temperature depending on body surface friction heating and body nose shape.

NACA engineers John Stack and Eastman Jacobs (1902–1987) in 1934 used schlieren photography to observe the presence of a shock wave on the top surface of an airfoil. Prior to this observation William J. Rankine (1820–1872) devised an equation for calculating the airflow across shock waves in 1870. This has now been replaced by the Rankine-Hugoniot equation attributed to the independent studies of William Rankine and Pierre Hugoniot (1851–1887) on shockwave flow properties.

*See* **Blunt nose re-entry vehicle; Mach angle & cone; Critical Mach number; Fishtail shockwaves; Bow shock wave; Lambda shockwave; Incipient shockwave; Normal shockwave; Oblique shockwave.**

**Short-period mode oscillation.**      *See* **Short-term (or short-period)**

**Short-term (or short-period) oscillation** Depending on the aircraft's stability, the oscillations in the pitching plane can be of low or high frequency duration. A short-term oscillation is a desirable stability asset for all aircraft. The positive static stability is correctly balanced when the pitching oscillation is of a short-term duration. It consists of rapid changes in angle of attack as the plane pitches about the lateral axis, the motion lasting a few seconds for each mode. The short-term oscillation is related to the static margin.

It is also known as rapid incidence adjustment or short-period mode.

*See* **Stability; Phugoid motion.**

**Shroud** The shroud is that part of the rear, upper surface of the wing, which covers the flaps as found on a split flap system, or some Fowler flaps. When the flaps are extended and lowered a small gap remains between the flaps and the shroud. Through this gap, the airflow over the extended flaps is re-energized due to the Coanda effect.

*See* **Coanda effect.**



**Shrouded propeller** A shrouded propeller is one of short radius with an odd number of blades encased in a shroud. On an aircraft, it is known as a propulsor and on a helicopter, as a Fenestron (buried tail rotor).

*See* **Ducted fan.**



**Photo:** This Drone has a mid-section engine and shrouded prop. It is displayed in the Udvar-Hazy Centre, Washington, D. C.

**Sideslip** The sideslip is a condition of unbalanced, crossed-control, banked flight. A sideslip induces the aircraft to track either side of the heading while descending with an increase rate of descent.

The sideslip is achieved by throttling back the engine and banking to one side with opposite rudder applied to stop the yaw towards the lowered wing. Centripetal force, which makes the aircraft turn, is absent. A degree of heading control is available by altering the

amount of rudder input up to maximum rudder deflection when the aircraft will then turn towards the lower wing.

The greater the angle of bank, the greater is the rate of descent and more rudder input is required to keep the aircraft tracking in the same direction. The manoeuvre is common for pilots of non-flap equipped aircraft to increase the rate of descent when approaching to land. Sideslipping in most aircraft with flaps down is prohibited due to the flaps blanketing the tailplane decreasing its effectiveness.

**Sideslip angle** The sideslip angle is of importance being the primary reference when considering lateral and directional stability.

The angle is considered positive when the relative wind is displaced to the right of the aircraft's centreline.

The Greek letter Beta ( $\beta$ ) denotes the sideslip angle.

**Similarity parameters** Similarity parameters are ratios used by aerodynamicists to convert data from models in wind tunnel experiments to represent full-scale aircraft. For example, the Reynolds number is a similarity parameter for the viscosity of a fluid, and Mach number is the similarity parameter for compressibility.

**Simple flap** Simple flaps are generally found on basic light aircraft. The flap is hinged at its leading edge and consists of only one vane and no slots.

**Single-wedge airfoil** Single-wedge airfoils are mounted on high-supersonic aircraft to cope with the supersonic flow. They have sharp leading edges, blunt trailing edges and flat sides. The vertical tail of the North American Aviation X-15 research aircraft is single-wedge shaped.

**Skew wing** See Oblique wing.

**Skin friction coefficient** The skin friction drag coefficient in an incompressible, viscous, laminar flow is defined as follows:

$$C_F = \frac{\tau_w}{\frac{1}{2} \rho U_m^2}$$

Where  $\tau_w$  = shearing stress at surface

$\frac{1}{2}$  = a constant

$\rho$  = air density

$U_m^2$  = mean flow velocity.

A variation in the skin friction coefficient occurs with changes in Mach number, Reynolds number and the boundary layer characteristics.

See **Skin friction drag**.

**Skin friction drag** Skin friction, a part of the parasite drag, contributes a major portion of the total drag at low angles of attack. This is shown on **Diagram 21, Drag Divergence Curve**, where the parasite drag curve slope remains almost horizontal due to very little change in skin friction with increasing angle of attack. When the angle of attack increases towards maximum, form drag predominates greatly with an increase in the slope of the parasite drag curve.

Skin friction drag is due to five factors, which cause resistance in the boundary layer airflow. These are listed below:

- shear stress; the shearing effect of the air molecules in the boundary layer's viscous airflow, causing resistance
- the wing's shape; skin friction is reduced when the wing's point of maximum camber is further aft
- air speed; increasing air speed increases the skin friction, a major factor in high Mach number flight
- surface roughness; increased skin friction is caused by a rough surface, with rivets, raindrops and ice, etc

- the aircraft's wetted area; the greater the aircraft's surface the greater will be the skin friction.

Charles-Augustin de Coulomb (1736–1806) a French scientist, introduced the Laws of Friction in the late 1800s, during his studies of tribology. Theodore Theodorsen (1897–1978) of NACA carried out further detailed research into skin friction in the subsonic, transonic and supersonic speed range. It is also known as surface friction drag and Reynolds stress.

See **Drag**.

**Slab tailplane** A slab tailplane consists of a single-pivot primary control surface to control the aircraft in the pitching plane. It is an all-moving surface devoid of elevators and fixed tailplane. They are normally found on high-speed jet transport aircraft and jet combat aircraft.

It is also referred to as a horizontal stabilizer by US pilots.

See **All-flying tailplane; Stabilator**.

**Slats** Slats are small auxiliary airfoils mounted on the leading edge of the main wing.

The slats move downward and forward into position to extend the wing's chord, creating more wing area and thus more lift and therefore a shorter take-off or landing distance. They also increase the wing's upper surface camber and direct an increased volume of airflow between the slat and the wing's leading edge, which reinforces the energy in the airflow over the wing at high angles of attack to increase the maximum lift coefficient and delay the stall onset.

Slats may be held open or closed by springs and extend automatically (automatic slat) due to the action of the low-pressure area (leading edge suction) at high angles of attack. This type of slat was used on the German Messerschmitt Me 109 fighter for example. The Moraine Saulnier MS-880 Rallye, single-engine light

aircraft has leading edge slats, which extend automatically due to leading edge suction when speed is reduced in flight to about 75 Knots. They are held in the extended position by springs at low speed and are retracted by increased dynamic air pressure with increasing air speed and reduced angles of attack. An alternative method is manual extension by the pilot when required. On larger aircraft, the leading edge slats extend with initial trailing-edge flap extension.

Leading edge slats form a slot at the leading edge as opposed to Krueger flaps, which have no slot, as found on some Boeing jet transports.

*See **Slots; Krueger flaps; Leading edge devices (LEX).***



**Photo:** The Curtiss O-52 Owl has full span leading edge slats. This example is located in the Pima Air & Space Museum, Tucson, Arizona.

**Slender wings** Three types of wing planform fall under the heading of slender wings. These are the delta wing, the Gothic wing and the ogee (or ogive) wing as found on Concorde. The wing has a low aspect ratio of long chord and short span, which keeps the entire wing inside the Mach cone emanating from the nose cone. It is mounted on high-speed or supersonic aircraft.

**Slew wing** *See Oblique wing.*

**Slip** The term slip refers to the difference between the advance per revolution of a propeller and its geometric pitch.

Slip can also be described as the difference between the effective pitch and the geometric pitch or, the difference between the slipstream velocity behind the prop and the aircraft's velocity, expressed as a percentage of the aircraft's speed. Slip is a variable.

When the advance per rev equals the experimental pitch, the angle of attack is slightly negative and produces zero thrust. However, in normal operating conditions, the angle of attack is around positive three degrees and the advance per rev is considerably less than the geometric pitch. **Diagram 37, Geometric Pitch**, shows the advance per rev (B-C) plus the slip (C-D) is equal to the geometric pitch (B-D). For the propeller to provide maximum thrust and efficiency, and because air is not a solid medium, slip must be present. Maximum efficiency is only obtained when the slip, expressed as a percentage is around 30% of the length of the geometric pitch. In other words, slip is the difference between the actual distance the prop travels forward in one revolution (B-C or effective pitch) and the distance the prop would theoretically travel in one revolution if its advance per rev were equal to the geometric pitch (B-D). Slip is a percentage of distance.

*See Diagram 68, Propeller Pitch.*

**Slip flow** Slip flow occurs at hypersonic speeds where the boundary layer does not become stationary at the wing's surface, but slides over it due to molecular shearing of the airflow.

This phenomenon was discovered by James Clark Maxwell.

**Slip function** The slip function is the ratio of the aircraft's velocity (TAS) to the product of propeller RPM times the diameter ( $V/nd$ ).

*See Advance/diameter ratio; Effective pitch ratio.*

**Slipstream** The airflow in the immediate vicinity flowing over the aircraft is the slipstream. The slipstream immediately behind the propeller is correctly known as propwash.

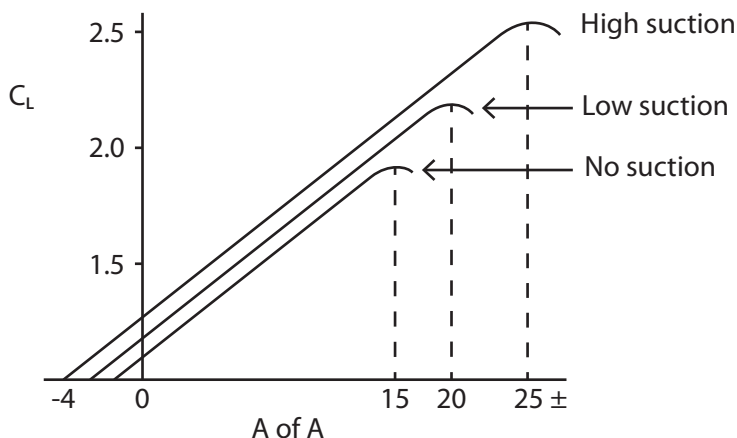
**Slipstream velocity** The slipstream velocity is the common, but erroneous name for propwash. The actual slipstream velocity is the speed of the total airflow over the entire aircraft during flight. (The propwash is the portion of the airflow directly behind the propeller).

*See Propwash-thrust.*

**Slots** The slot – a form of boundary layer control – is a small gap located between the fixed or retractable slat and the wing's leading edge.

The purpose of the slot is to re-energize and direct airflow from below the leading edge to the upper surface. The pressure differential between the lower and upper surface of the wing induces air to flow at a higher speed through the slot. The increased speed of flow over the upper surface maintains a laminar flow over a greater portion of the chord delaying the change to a turbulent boundary layer thus increasing the maximum lift coefficient at a higher angle of attack and therefore, delaying the stall. The re-energized flow over the upper surface due to the venturi effect also

helps to straighten out the flow and reduce the spanwise trend towards the tips and thus delay or prevent wingtip stalling. The lift can be increased by up to 20% due to the slot effect with very little increase in drag and only a small pitch change.



**Diagram 75, Slots & Lift Coefficient**

Double or triple slotted flaps are separated by slots, which have the same effect as a leading edge slot where they increase and energize the airflow over the individual flap vanes. Other control surfaces may also have a slot at their leading edge. The slot may be inverted on some tailplanes; the tailplane normally provides a down force to balance the main wings.

The Handley-Page Aircraft Company Ltd, developed and patented the leading edge slot in 1919, which were invented by the German, Dr. Gustave Victor Lachmann (1896–1966) in 1916. Slots were flight tested on an Airco (DeHavilland) D.H.9 World War I fighter at the Royal Aircraft Establishment, Farnborough, UK in 1920.

**See Slats; Krueger flaps; Powered slat; Leading edge devices (LEX); Handley-Page, Sir F.; Lachmann, G. V. Dr.**





**Photo:** The leading edge slots on the Culver Cadet are of the fixed slot type. This aircraft is on display at the National Aeronautical & Space Museum in Santiago, Chile.

**Slotted airfoil** Two types of slotted airfoils exist. One type has the leading edge slot permanently fixed in position on the wing's leading edge. The second type is a gap formed by an extendable slat at the leading edge when deployed into position to form the slotted airfoil.

*See* **Slots; Slat; Leading edge devices (LEX).**

**Slotted aileron** A slotted aileron has a slot located permanently along its leading edge to improve airflow over the aileron's upper surface. This is in direct contrast to a sealed control gap where the airflow is prevented from flowing through the gap to prevent drag.

**Slotted flaps** The slotted flap invention was an improvement over the plain or split flap, where a greater lift coefficient is produced for not much increase in drag. As with slotted ailerons, the slotted flap increases the velocity and re-energizes the airflow in the boundary layer directed over the flap's upper surface. This is due to a fresh boundary layer starting at the slot ahead of the flap's leading edge rather than using the turbulent boundary layer flowing directly off the wing. The airflow over the entire wing is improved with a reduced adverse pressure gradient, which delays flow separation and leads to a higher maximum lift coefficient.

Most flap systems used on today's modern aircraft have at least one slot, more often than not, two or three slots are incorporated in the system to raise the maximum lift coefficient.

The Handley Page Aircraft Company experimented with slotted flaps during the 1920s. A German engineer by the name of O. Mader working for Junkers also developed a type of flap with a long, wide slot between the wing's trailing edge and the airfoil attached to it, known as the Junkers flap.

*See* **Junkers flaps; Flaps; Foreflap.**

**Small shock stand-off distance** In supersonic flight, the bow shockwave on a blunt nose aircraft is designed to stand-off a short distance ahead of the nose. With increasing Mach numbers, the stand-off distance reduces due to the decreased volume aft of the shockwave in accordance with the conservation of mass. Dissociation of various gases occurs with very high Mach number air flow through the bow shockwave.

See *Diagram 74, Shockwaves; Viscous flow.*

**Smeaton coefficient** The Smeaton coefficient was an original lift formula developed in 1759 by John Smeaton (1724–1792) an English civil engineer. The formula involved the lift force on a flat plate acting normal to the airstream. The coefficient given by John Smeaton as 0.005 was later corrected by the Wright brothers to 0.0033, based on the number of pounds of drag acting on a flat plate at 1 MPH. The formula is given as follows:

$$\text{Lift} = k V^2 A C_L$$

Where  $k$  = Smeaton coefficient of 0.0033

$V^2$  = velocity squared

$A$  = wing area in square feet

$C_L$  = lift coefficient.

**Solidity** Solidity is the ratio of total blade area to the total propeller or rotor disc area.

Each propeller (or rotor) has a maximum limit to the amount of thrust it can produce. If the aircraft designer requires more thrust, then a propeller of greater solidity will be required. Solidity, the ratio of total blade area to the total propeller or rotor disc area can be found from the following formula:

$$\text{Solidity} = S/\pi R^2$$

Where  $S$  = total blade area

$\pi$  = 3.14

$R^2$  = prop disc area.



**Photo:** A pair of six-blade composite propellers powers the British Aerospace BAe ATP turboprop airliner.

A two-blade propeller has a solidity of about 0.08 increasing up to around 0.16 for a four-blade propeller. Solidity and therefore thrust, can be increased by using a propeller with the following:

- a greater diameter
- a greater number of blades
- or wider chord blades.

Over the years, the number of blades that could be effectively used gradually increased with improved propeller design. Up until just a few years ago, six blades was the maximum number that could be mounted on one propeller. Now that number has increased to eight. The Northrop Grumman Hawkeye 2000 EWACS twin-turboprop aircraft is one example which sports a pair of eight-blade scimitar shaped propellers.

**Sonic barrier** *See Sound barrier.*

**Sonic boom** Sonic booms are caused by the pressure waves emanating from a supersonic aircraft.

An aircraft flying at supersonic speeds forms a Mach cone at the nose of the aircraft and a fishtail shockwave at the trailing edge. These two shockwaves (the Mach cone is a form of shockwave) reach the ground almost simultaneously causing an audible change in air pressure. This is heard as two successive sonic booms by an observer below the aircraft's flight path after the plane has passed overhead. However, the two booms are so close together that the boom is usually heard as one loud crack when nearby or as a thump at a distance. The longer the aircraft fuselage, the greater is the separation and timing of the booms arrival at an observer's location. For example, Concorde's two booms are a quarter of a second apart but are much closer for fighter type aircraft.

Although shockwaves are heard as sonic booms by observers on the ground, they are not strictly the same as ordinary sound waves initially; they change over a distance due to heating, lose energy and become like conventional sound waves.

Shockwaves may refract back to the earth's surface from the upper atmosphere producing a quieter, secondary sonic boom at less intensity and which may not be noticed by anyone. With increasing distance away from the aircraft's track, the boom becomes quieter. If the aircraft is accelerating, decelerating, or manoeuvring, the boom becomes a focused or louder boom, up to 200 decibels. The size, weight, and length of the aircraft determine the severity of the sonic boom.

See **Overpressure; Blunt nose re-entry vehicle; Refracted boom.**

**Sonic lines** At velocities above Mach 1.0, a region of supersonic flow is separated from a subsonic flow region by a sonic line. For example, *Diagram 8, Detached Normal Shockwave*, shows the sonic lines separating the subsonic flow behind the detached shockwave and the supersonic flow to the sides of the blunt nose. The sonic line depicts an indistinct boundary, as opposed to a definite boundary produced by a shockwave.

**Sonic waves** At subsonic flight speed, pressure, or sonic waves move ahead to 'warn' the air of the approaching aircraft. As the aircraft approaches and exceeds Mach 1.0, the air has no warning time of the approaching aircraft, and so the sonic waves 'pile up on each other' and a shockwave is formed.

*See Diagram 57, Mach Angle & Cone.*

**Sound barrier** Due to the high drag rise in the transonic region approaching Mach 1.0 (alias the speed of sound) it was wrongly assumed a sound barrier existed and prevented aircraft from reaching Mach 1.0. It was not until engines of greater power became available that aircraft were able to penetrate the so-called sound barrier. It was, and still is the increase in aerodynamic drag due to compressibility in the transonic region that is a hindrance to exceeding the speed of sound. The speed of sound just happens to lie in the middle of the transonic region and sound in itself is not a barrier, but the aerodynamic drag is! Reference to *Diagram 21, Drag Divergence Curve*, shows how the drag increases close to the speed of sound.

The term sound barrier (the barrier does not exist) was coined back in the early days of research into supersonic flight. The British press introduced the term after misinterpreting a remark made by British aerodynamicist, Dr. W. F. Hilton at the National Physics Laboratory in London, England, in 1935.

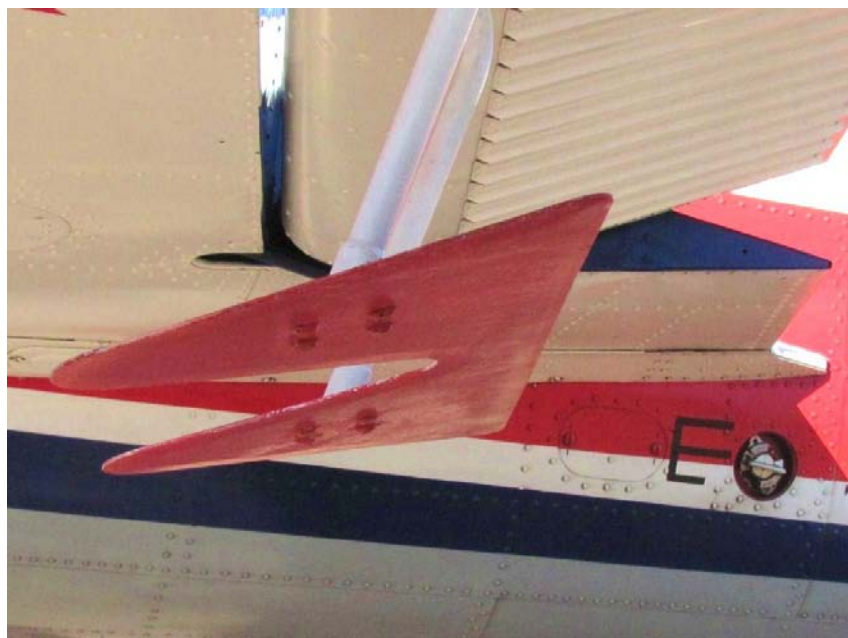
The first aircraft to exceed the speed of sound was the rocket-powered Bell X-1 piloted by the now famous, General Charles (Chuck) Yeager (1923–20?) on 14 October 1947.

The term ‘compressibility barrier’ is a more exact definition than the term sound barrier, or sonic barrier, which should no longer be used – it should be relegated to the archaic file.

It is also known as the sonic barrier.

See **Compressibility barrier; Detached shockwave; Speed of sound (a)**.

**Spades** Spades are horizontal, pylon mounted plates mounted to the aileron’s under surface. They provide extra surface area forward of the hinge line axis to lighten the aileron control feel using aerodynamic balance; they are generally found on aerobatic aircraft. When up-aileron is applied, the spades angle downwards into the airflow, which applies pressure to the upper surface of the spade and hence, pushes up the aileron, and vice versa on the opposite down-going aileron.



**Photo:** A spade mounted to the aileron on a Yak 55M aerobatic aircraft.

**Span** An aircraft's aerodynamic wingspan is the total lateral distance from one wingtip to the other.

This definition must be modified when quoting the span for variable geometry (swing-wing) aircraft. The span may, or may not include winglets or wingtip fuel tanks. The term span could also refer to the structural span, which is the physical length of the wing. On a straight wing aircraft, the structural and aerodynamic spans are both the same but on a swept wing aircraft the structural span is longer.

It is also known as wingspan.

**Span efficiency factor** The wing span efficiency factor ( $e$ ) is a part of the induced drag coefficient formula.

According to Professor Ludwig Prandtl's (1880–1953) lifting line theory for a wing with a relatively small sweep angle, the induced drag formula is:

$$C_{Di} = \frac{C_L^2}{\pi AR e}$$

Where:  $C_{Di}$  = induced drag coefficient

$C_L^2$  = lift coefficient squared

$\pi$  = 3.14

AR = aspect ratio

$e$  = span efficiency factor.

The value of  $e$  for most wings, is determined by the shape of the lift distribution in accordance with the lifting line theory, and will be a maximum of one for an elliptical shaped wing (but generally closer to 0.9) reducing to 0.7 for a rectangular wing, depending on the angle of attack. The efficiency factor for an elliptical wing is omitted from the formula when its value is one.

It is also known as the Oswald's efficiency factor.

**See Induced drag coefficient.**



**Span loading** The span loading is defined as the ratio of the aircraft's weight divided by the square of the wingspan.

The span loading affects the aircraft's rate of climb and service ceiling. Thrust horsepower required is proportional to the square of the span loading and therefore a low span loading is a requirement for high performance aircraft.

$$\text{Span loading} = \text{weight/span squared } (W/b^2).$$

See **Elliptical lift distribution**.

**Span loading parameter** The span loading parameter describes the aerodynamic load at each spanwise station across the span of the wing.

An elliptical lift distribution is the ideal span loading; however, sweptback wings may become more loaded at the wingtips than at the roots. This could result in the wingtip stalling prior to the wing roots, which is very undesirable for stability purposes.

$$\text{The span loading parameter} = \frac{\text{section lift coefficient times the chord (at that station)}}{\text{total wing lift coefficient times MAC.}}$$

The section lift coefficient is the lift coefficient acting on the chord at the chosen station across the span.

**Spanwise** The term spanwise refers to the lateral direction across the aircraft, from wingtip to wingtip.

*Compare with* **Span**.

**Spanwise axis** See **Feathering axis**.

**Spanwise lift distribution** The semi-elliptical plot of the wing's lift distribution on the upper surface and viewed from in front of the aircraft. A perfect elliptical lift distribution is preferred but the actual shape is governed by the wing planform.

*See Elliptical lift distribution; Lift-grading curve.*

**Spanwise pressure gradients** Reference to pressure gradients, usually infers a chordwise gradient. However, spanwise pressure gradient induces a lateral flow towards the tips. The decreased boundary layer energy flowing in the lateral spanwise pressure gradient exists mainly on swept back wings. A spanwise pressure gradient towards the wingtips has the following effects:

- less lift, reduced lateral and longitudinal stability, less aileron effectiveness and an increase in stall speed
- premature tip stalling causing an unstable pitch-up
- it requires the installation of vortilons and fences, etc.
- induces the formation of a rams horn vortex
- The airflow readily transitions from a laminar flow to a turbulent flow. This is the reason for flow fences on sweptback wings – to reduce spanwise flow.

*See Sweptback wings.*

**Spats** Spats are found on some single-engine light aircraft and are used to streamline the airflow over the fixed undercarriage wheels to reduce drag.

Spats are more commonly known as wheel fairings.

**Speed brakes** Speed brakes are used to dissipate the aircraft's energy by increasing its drag without affecting the wing lift. The speed brake's angle of deployment determines the increase in drag and therefore the airspeed reduction; lift is not affected.

Modern jet fighter aircraft such as the McDonnell Douglas F-15 Eagle use speed brakes. On the F-15, the speed brake is mounted on

the dorsal position between the two vertical fins. The British BAe 146 airliner and also the Blackburn Buccaneer (a two-seat Naval attack aircraft) are two examples with split air brakes known as diverters, forming the tail cone, which hinge outwards to provide drag.

See **Air brakes; Dive brakes; Diverters; Lift dumpers; Spoilers.**

**Speed bumps** A speed bump is attached to the wing's trailing edge on some high-speed aircraft. The streamlined bulge acts in the same manner as area rule to reduce drag.

See **Shock body; Whitcomb body; Küchemann carrots.**

**Speed of sound (a)** Sound waves are transmitted through the air at the speed of sound as small pressure waves. The pressure waves alternately compress and expand the air as they travel outward from the source, which causes the neighboring air to do the same. Compression and expansion of the sound waves are due to the air molecules vibrating about their mean position.

Temperature, density and pressure all decrease with increasing altitude and have an effect on the speed of sound, which varies directly with the ambient temperature and pressure and indirectly with air density. Because density is closely related to temperature, it is ignored in our calculations. It follows it is the temperature of the air of standard composition, which determines the actual speed of sound.

The sea level International Standard Atmosphere temperature is stated to be +15°C (288° Absolute) where the speed of sound is 661 Knots (1116.4 FPS) and varies in proportion to the square root of the Absolute temperature. The temperature in the stratosphere (above 36000 feet) is assumed to be at minus 56.5°C (or 216.65° Absolute) where the speed of sound reduces to 580 Knots (971 FPS). This is a decrease in the speed of sound of approximately 5 FPS per 1000 feet of altitude increase. Conversely, an increase in temperature above standard conditions (+15°C or 288° Absolute)

will increase the speed of sound, which can occur for example in a detached bow shockwave or in any compressed airflow, which gains heat.

The speed of sound is also known as the acoustic velocity, or Mach 1.0, after Dr. Ernst Mach (1838–1916) the Austrian philosopher and physicist. Mach number is the ratio of aircraft speed, or the propeller tip speed, to the speed of sound. This is expressed as a decimal number; i.e. Mach 0.5 equals half the speed of sound, etc.

The ratio of specific heats, the gas constant, and absolute temperature together define the speed of sound in Knots; this can be simplified by the following:

Speed of sound = constant (38.9) times  $\sqrt{\text{absolute temperature (Kelvin)}}$ .

Sir Isaac Newton (1642–1727) was the first person to make early calculations on the speed of sound, which were later corrected in 1789 by Pierre-Simon Laplace (1749–1827) when he included the air density in his calculations.

As an industry standard for all future reference purposes, NACA determined the exact speed of sound at sea level with a temperature of +15°C (288° Absolute) to be set at:

- 661 Knots
- 761.2 MPH
- 1116.4 FPS
- 340 m/s
- 1225 km/h

See **Compressibility barrier; Detached shockwave; Sound barrier.**

**Speed ratio** A particular propeller will normally be chosen from a family of propellers to suit the aircraft's design cruise speed. In order to produce the maximum amount of thrust for the least amount of drag or torque, each blade element is set to a different angle to ensure the optimum angle of attack is maintained. At the design point for a certain forward speed and propeller RPM (known as the speed ratio), the propeller will produce its maximum efficiency. This is the reason why the propeller blade is twisted. If the blades were not twisted, the blade root would be operating at a negative angle of attack while the propeller tip would be stalled.

The speed ratio is the ratio of the aircraft's forward speed to propeller RPM (TAS/RPM).

See **Propeller efficiency**.

**Speed stable/unstable** An aircraft is said to be speed stable when flying above the minimum drag speed and speed unstable when flying below that speed.

Reference to *Diagram 20, Total Drag Curves*, shows the total drag increases above the minimum drag speed, where parasite drag increases and induced drag decreases with increasing speed. The reverse happens below the minimum drag speed. The total drag increases above and below the  $V_Y$  minimum drag speed.

Consider an aircraft flying at constant power above the minimum drag speed and the aircraft experiences an increase in speed. Due to the increased total drag, the aircraft will decelerate back to its trim speed. The converse happens if the aircraft's speed decreases; the drag will have reduced to a value less than the thrust and so the aircraft will accelerate until thrust and drag are once again equal.

At speeds below the minimum drag speed, the aircraft is speed unstable due to the total drag increasing with decreasing speed. A speed reduction while at constant power below  $V_Y$  will slow the aircraft to a speed where the drag is greater, which will slow the aircraft further – a speed unstable situation.

See *Diagram 20, Total Drag Curves*.

**Spinning** The spin is a loss of control flight-manoeuvre involving a combined stall and yaw condition.

During the spin, the angle of attack is beyond the critical angle, at least on one wing causing asymmetric lift. The aircraft is also yawing, pitching and rolling. This results in a steep spiral descent with the aircraft rotating about its centre of gravity. The spin may be flat, steep, or inverted, with a steep spin being the preferred choice for a quicker and safer recovery. The spin manoeuvre has three distinct stages – the incipient or entry stage, the in-spin condition and the recovery stage.

The entry stage involves an asymmetric stall due to a high angle of attack on one wing. The increased drag of the stalled wing causes the aircraft to yaw towards that wing. The rising wing being at a lower angle of attack produces less drag and greater lift and so the aircraft rolls. The fuselage can also have an adverse reaction in causing the aircraft to spin. Accidental spins are commonly caused by improper manoeuvres, for example, during low-air-speed steep turns at low altitude with too much rudder input or poorly performed aerobatics.

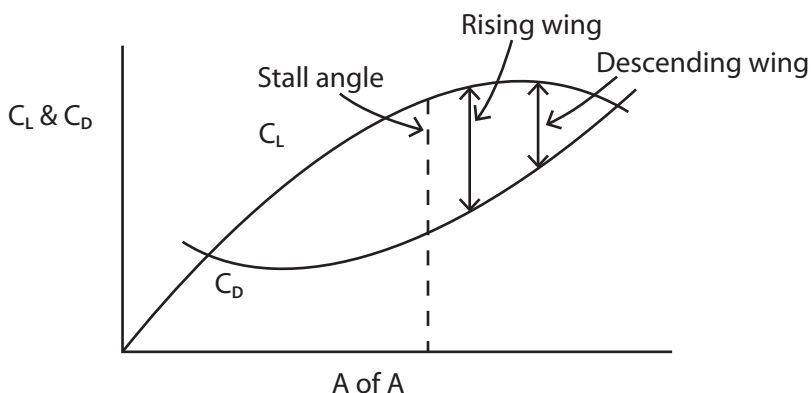
Following about one to three turns the spin is fully developed. A constant rate of rotation and rate of descent becomes established during the inspin stage. The nose attitude is well down in a steep spin with a relatively large spin radius. A nearly horizontal attitude results in a flat spin where the spin radius is much smaller, with the aircraft yawing about its OZ normal axis.

The spin is maintained by a combination of aerodynamic and centrifugal forces. Due to the low nose attitude, the centre of pressure is tilted forward and on a stalled wing, it moves aft along the chord line. The resulting couple between the centre of pressure and centre of gravity causes the nose attitude to lower further. Opposing this nose down couple are the centrifugal forces acting on each end of the fuselage providing a couple that tends to raise the nose. The balance between the two couples determines the final nose attitude during the spin. A multi-engine aircraft also has

the centrifugal force of the engine's weight adding to the nose up attitude; these types of aircraft inherently enter into a flat spin.

Most spins require recovery action by the pilot; aircraft that are more docile will recover to a steep spiral descent with relaxed control pressures – the Victa Airtourer is one that comes to mind. During recovery, the rate of rotation decreases, the wings become un-stalled with the nose in a low attitude and air speed increasing towards normal flying speed. The steep spin is the easiest spin to recover from, due to the wings being at an angle of attack at or just beyond the stall where the controls retain some effectiveness. The flat spin is more difficult to recover from due to the wings being deeper in the stall with the high nose attitude and the controls being less effective. The inverted spin is also difficult to recover from due to the aircraft being upside-down causing confusion for the pilot and requiring opposite control inputs to a normal spin recovery. It may be impossible to recover from some flat and inverted spins. It all depends on the aerodynamic forces and inertial moments acting on the aircraft during the spin – and the pilot's flying skills!

**Diagram 76, Spinning**, shows the rising wing has greater lift coefficient and lower drag coefficient than the descending wing which is deeper in a stalled condition.



**Diagram 76, Spinning**

Sidney B. Gates (1893–1973) an aerodynamicist at the Royal Aircraft Establishment, Farnborough, UK, was instrumental in research of spin manoeuvres and recovery techniques.

It is also known as autorotation. See flat spin, inverted spin and normal spin, precision spin and transition spin.

**Spin tunnel** A spin tunnel is one built vertically with the air flowing upwards. The model under test is mounted in a nose down attitude, as it would be in free flight, from where the spinning characteristics can be studied.

**Spiral dive** A spiral dive appears similar to a normal spin except in the spiral dive the wings are not stalled, the aircraft is under control and the air speed increases readily if not checked by the pilot.

**Spiral divergence** Spiral divergence is an unstable flight condition. Following a yaw, the aircraft rolls and enters a slow spiral dive, which quickly develops into a steeper nose-down and tighter spiral dive, rapidly losing altitude with air speed increasing. The flight upset can easily be corrected by a pilot flying visually but can be very disorientating for an inexperienced pilot flying on instruments.

Spiral divergence is the result of strong static directional stability combined with weak dihedral effect. The former tends to turn the aircraft's nose into the relative wind to restore the aircraft to lateral level flight; the dihedral effect has less effect on the aircraft's stability and recovery from the spiral divergence.

It is also known as a graveyard spiral.





**Photo:** The Seversky P-35A pursuit aircraft, used by the USAAF, is shown with its split flaps extended. This aircraft is on display in the National Museum of the USAF, in Dayton, Ohio.

**Spiral mode stability** An aircraft with spiral mode stability is one that tends to recover to straight and level flight from a banked attitude.

It is a desired flight characteristic for an aircraft to revert to straight and level flight following the release of controls during a banked attitude. Most aircraft have spiral mode stability at very small bank angles; however, as the bank angle increases spiral mode stability reduces, the aircraft becomes unstable to the point of spiral divergence. The change from spiral stability to spiral instability varies between aircraft types but is easily controlled by pilot input. Spiral mode stability is the opposite to spiral divergence.

It is also known as spiral mode, or roll stability, which is an ambiguous term.

**Split flaps** Split flaps form a part of the wing's lower trailing edge surface and when deployed while the wing's upper surface remains intact. The flaps have little effect on increasing the maximum lift coefficient itself – maybe up to 17% maximum. However, the substantial increase in drag is due to the turbulent airflow behind the lowered flap.

Orville Wright (1871–1948) and James M. Jacobs in 1920 invented the split flap and patented it in 1922. They were first used, for example, on the Northrop Gamma, Lockheed 10A and Orion, Douglas DC-1, DC-2, DC-3 and Curtiss C-46 Commando.

**Spoilerons** Spoilerons are primary flight controls used for roll control in full-time operation. Their location on the wing's upper surface frees up the trailing edge for flaps. Spoilerons can be used differentially to achieve the effects of ailerons by reducing the lift



**Photo:** The Northrop P-61 Black Widow uses spoilerons for roll control. This Black Widow was photographed at the Udvar-Hazy Centre in Washington, USA.

of one wing but unlike ailerons not increasing the lift of the other wing. A raised spoileron also increases the drag on the wing that is lowered, counteracting adverse yaw. It is used in place of ailerons, thus reducing the number of control surfaces required. A spoiler is pro-yaw as opposed to ailerons, which produce adverse yaw.

Northrop pioneered the use of spoilerons when they built the WWII Northrop P-61 Black Widow night fighter, which made its first flight on 21 May 1942. A few years later, the Boeing B52 Stratofortress jet bomber entered service with the USAF, equipped with spoilerons in lieu of ailerons. The B-52's high aspect ratio wing is so flexible that the ailerons would cause too much distortion and possible aileron reversal. The Mitsubishi MU-2 twin-turboprop is also equipped with spoilerons. The MU-2 has a wing area of 178 sq. ft., only a fraction greater than a Cessna 180 but carrying about four times the weight, which means a very high wing loading and hence, a high landing speed. In order to keep the landing speed down, double-slotted flaps adorn the entire length of the wing's trailing edge leaving no room for ailerons, hence the need for spoilerons.

**Spoilers** Spoilers are multi function, secondary control surfaces, operating differentially or in unison. They are located on the wing's upper surface and hinged at their leading edge to be raised up into the airflow to spoil the lift. Flight spoilers, also known as air brakes, are used symmetrically to increase drag and reduce speed, and/or increase the rate of descent. The spoilers work by disturbing the airflow to create turbulence and reducing circulation lift behind them on that part of the wing. Spoiler activation reduces the lift/drag ratio due to increased form drag; to maintain the required lift the angle of attack must be increased. They also produce a higher stall speed.

Sweptback wings on high-speed jet aircraft can suffer from adverse effects of ailerons, which can cause adverse yaw, wing flexing, control reversal, and become less effective at high Mach numbers. Roll augmentation spoilers on the other hand, are a

great advantage in counteracting adverse aileron yaw, providing roll control, and drag when required. They have the disadvantages of not providing any control feel. In addition, they cause vibration and noise in the disturbed airflow behind them, which can be disconcerting for some passengers.



**Photo:** The spoilers in the raised position on the port wing of an Airbus A320, in flight.

After pre-arming by the flight crew, a micro-switch on the undercarriage activates the spoilers to full extension, in unison, to act as lift dumpers and destroy lift as the aircraft touches down on landing. This places the aircraft's weight on the wheels to assist braking. Ground spoilers usually deflect to a greater angle than flight spoilers. Flight and ground spoilers are both the same control surface, taking their names from where they are used, in flight or on the ground respectively.

Spoilers are found mostly on sailplanes and large jet aircraft. Ground spoilers, are also known as auto spoilers or lift dumpers.

*See* **Air brakes; Speed brakes; Dive brakes; Diverters; Lift dumpers.**

**Spring tab** The spring tab (or servo tab) is a form of trim tab hinged at the trailing edge of a control surface designed to relieve the force required by the pilot to move the control surface; as the name implies, it is spring loaded. A small control up-deflection input by the pilot results in the control surface moving readily. Larger control inputs are assisted by the spring servo tab, which assists movement of the control surface.

**Square stall** A square stall (or straight stall) occurs when both main wings stall simultaneously with the aircraft in a wings level attitude. The nose pitches down without any yaw being involved. This is an ideal characteristic due to the absence of any yaw and the aircraft is unlikely to enter an incipient spin.

**Stabilator** A stabilator is a horizontal tailplane combining the stabilizer and elevator moving as one unit for aircraft control and stability in the pitching plane.

The advantages of a stabilator over a conventional tailplane are:

- its size is generally smaller, with less moving parts and is therefore lighter in weight yet still as effective and more efficient
- it is more rigid and less prone to flutter
- at high-speed, trim drag is less
- full elevator range available for greater pitch control
- Ideal for large CG and speed range, and large pitching moments due to configuration changes.

The stabilator is devoid of an elevator but has an anti-servo tab for trimming purposes attached to its trailing edge. When the trailing edge of the stabilator is raised or lowered as the case may be, the tab works to re-align itself with the stabilator and return it to an in-stream position. When it is aligned, the aircraft achieves an attitude where zero stick force exists. Stabilators pivot about their aerodynamic centre, located at the 25% MAC position, to provide a constant pitching moment at all angles of attack, thus requiring less pilot input force to move it. Canards are also a

form of stabilator mounted the forward fuselage. A stabilator on fighter aircraft may also provide roll control; they are the n termed tailerons or rolling tailplanes.

The pitch-trimming anti-servo tab's movement produces stabilator control feel on light aircraft, the greater the control movement – the heavier is the control feel. The Piper PA 28 Cherokee line of light aircraft and the Cessna 177 Cardinal have a stabilator. Modern fighter aircraft with swept or swing-wing design also employ stabilators. The Lockheed L-1011 jet transport also has a stabilator as opposed to most jet airliners, which have adjustable stabilizers.

The first aircraft to employ a stabilator was the Miles M.52 research aircraft, which first flew on 8 October 1947. Only one M.52 was built before the project was cancelled. The Bell X-1 supersonic aircraft of Chuck Yeager fame was the next aircraft to have a stabilator, followed by the North American Aviation F-86



**Photo:** North American F-86D Sabre shown here makes use of a stabilator for pitch control. This Sabre is located in the Pima Air & Space Museum in Tucson, AR, USA.

Sabre. The F-86E model had a powered stabilator installed with a linked elevator and later models had the stabilator without the elevators.

It is also known as a slab tailplane, all-moving tailplane or all-flying tailplane.

See **Stabilizers**.

**Stability** The centre of gravity location plays a very important part in longitudinal stability. For maximum stability, it is a requirement for the centre of gravity to be located slightly forward of the centre of pressure. If the centre of gravity is allowed to move further and further towards and then past the aft centre of gravity limit, the aircraft's static stability will gradually reduce to zero and then become negative.

Two types of stability motion exist – static and dynamic stability. Static stability and control forces are closely related and determine the aircraft's flying quality; increase static stability and control forces become heavier with less controllability and vice versa. The lower limits of controllability determine the maximum limit of stability. Likewise, the upper limits of controllability determine the lowest acceptable stability limit. Hence, stability and controllability are close relatives. Modern fighter aircraft are highly manoeuvrable with light control forces but are less stable, as opposed to a jet transport aircraft, which has greater stability but heavier control-feel making it less manoeuvrable. A modern light aircraft's stability and controllability would lie somewhere between these two extremes.

Static stability describes the aircraft's *tendency* to return to its trim speed after a disturbance. All aircraft can have positive, neutral, or negative static stability depending on the centre of gravity location. However, the only acceptable condition for safe flight is positive static stability, which is achieved with the centre of gravity remaining within its prescribed limits. The aircraft is said to have aperiodic stability, which is strongly damped and causes

the aircraft to return to its trimmed speed without any oscillations. An aircraft with positive static stability after a small displacement from its flight path, for whatever reason, the initial tendency is to return to its original trim speed. This is due to changes in the pitching moments about the fuselage, wings and empennage, etc and their associated forces. The only acceptable condition for safe flight is a short-term oscillation in pitch, also known as short period mode or rapid incidence adjustment.

Dynamic stability follows static stability; it describes the success of the static stability's restoring moment or lack of it. Dynamic stability is the term used to describe the *nature* of the plane's long term, phugoid oscillation around the OY lateral axis after it has returned to its original trim speed. Depending on the velocity of the aircraft, the resulting sinusoidal motion (like a sine wave) may take between 20 and 40 seconds or longer to complete one oscillation as the aircraft seeks to maintain straight and level flight. The time taken for the oscillations to damp out is proportional to the speed of the aircraft. The three degrees of dynamic stability are positive, negative or neutral as shown by **Diagram 77, Dynamic Stability**.

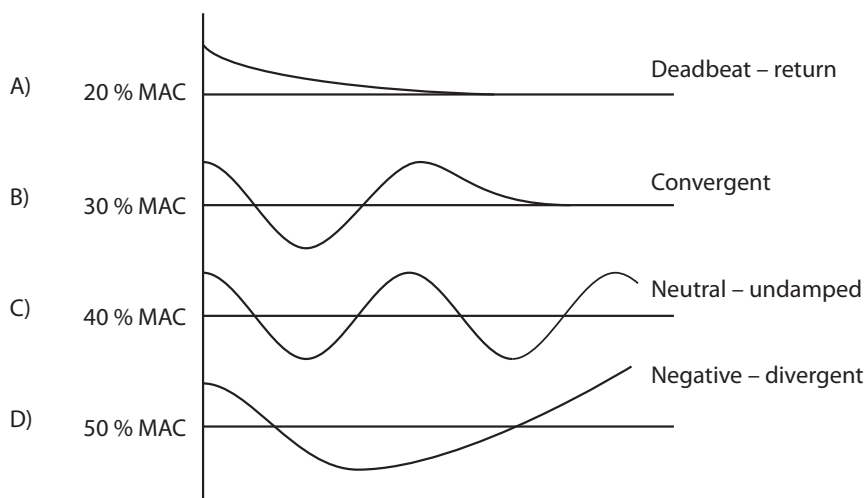
Dynamic stability gradually reduces as the centre of gravity moves rearwards up to around 30% MAC. In its most acceptable condition the aircraft will, after a disturbance from its flight path, return to straight and level flight with reducing oscillation. The oscillation is said to subside or to be deadbeat or damped and is the most ideal of all stability conditions where the aircraft returns to its original attitude without any oscillation, as shown by **Diagram 77A**. Nevertheless, positive static and positive dynamic stability is more common with a damped oscillation; the aircraft experiences several oscillations of decreasing magnitude, shown by **Diagram 77B**. Moving the centre of gravity aft will incur greater oscillations before the upset is damped out; approximately 40% MAC represents positive static stability and neutral dynamic stability, shown by **Diagram 77C**. With the centre of gravity well aft of the rear limit at about 50% MAC, the aircraft will be unstable. In this



condition, the oscillations diverge further and further from the required attitude, shown by **Diagram 77D**. The aircraft is statically stable but it has negative, divergent dynamic stability.

The degree of stability also depends on whether the controls are stick-free or stick-fixed. Airplanes and helicopters are more stable when stick-fixed – the pilot holds the flight controls steady. Helicopters are usually statically stable but dynamically unstable in the longitudinal, lateral, and yawing planes and especially in the hover. Longitudinal stability improves with increasing forward speed due to the stabilizer taking effect.

The reduced air density at altitude is a disadvantage to high-flying jet airplanes. The reduced air density manifests itself as reduced stability, reduced dynamic forces, which reduce the restoring moments and aerodynamic damping. The plane becomes less manoeuvrable with reduced control response. Dutch roll can make its presence felt, hence the need for yaw dampers.



**Diagram 77, Dynamic Stability**

Sir George Cayley (1773–1857) was the first person to have recorded any reference to aircraft stability and control. Alphonse

Penaud later expanded on this circa 1871, when he confirmed that a tailplane was required for stability purposes.

*See Lateral static stability; Lateral dynamic stability; Dynamic stability; Short-term (or short-period) oscillation; Phugoid motion; Positive static stability; Stick-free; Stick-fixed; Diagram 12, Centre of Gravity Range.*

**Stability margin** Stability margin is an alternate name for static margin.

*See Static margin; Diagram 63, Pounds of Pull V. CG Percentage MAC; Diagram 78, Static Stability V. Angle of Attack.*

**Stability of canard wings** A rearwards stick movement raises the aircraft's nose and the canard's net lift force moves forward and increases, resulting in the desired nose-up pitch movement. The aircraft's pitch moment then returns to zero, and the total lift returns to its original value. The increase in main wing lift is less than the canard's lift increase. The speed will of course, decrease with a pitch-up.

**Stabilizers** The stabilizer on an airplane refers to the vertical fin and horizontal tailplane. Its purpose is to provide stability and control in the pitching plane around the OY lateral axis (horizontal tailplane) and stability and control in the yawing plane around the OZ vertical axis (vertical fin).

The horizontal tailplane may be mounted on the airplane's fuselage below the vertical fin or as a cruciform, roughly mid-way up the vertical fin or as a T-tail mounted atop the fin itself. The horizontal tailplane usually provides a download in normal cruising flight to balance the couple produced by the vector of weight and the lift vector on the main wing. The vertical fin is always of symmetrical form and may be canted off-centre to allow

for the effect of the helical slipstream from the propeller on single-engine aircraft.

The Americans when referring to the aircraft's tailplane favor the term stabilizer.

See **Slab tailplane; Inverted stabilizer; Stabilator; Tailplane.**

**Stabilizer stick force** The stabilizer stick force is zero when the aircraft is flying at its trimmed speed with zero trim moment. However, with greater movement of the stabilizer, more force (pounds of pull) is required by the pilot to move it.

It is also known as pounds of pull.

**Stable airfoil** An airfoil is considered stable when the centre of pressure has very little travel with changes in angle of attack.

**Stabilons** Stabilons are short-chord panels streamlined with the airflow, located on the lower fuselage surface below the tailplane.

They have the combined effect during flight at normal cruise angles of attack where they aid the vertical fin in directional stability to damp out any yaw tendencies. They also come into effect with increasing nose-up attitude; their increasing angle of attack provides a lifting force under the tail to force the aircraft's nose to pitch downwards. In effect, they are an anti-stall device. One aircraft that comes to mind with stabilons is the Beech 1900D commuter.

**Stagger** Stagger is the difference in distance (measured along the OX longitudinal axis) of the leading edge of the upper and lower wings of a biplane. The wing is said have positive stagger if the upper wing leading edge is forward of the lower wing – the most common arrangement. The reverse arrangement with the upper wing set aft of the lower wing is negative stagger, but is less common. The Beech Model 17 Staggerwing built from 1932 to 1945, is a good example of negative stagger.

Some early biplanes had no stagger – the top wing is mounted vertically above the lower wing, known as an orthogonal biplane. The centres of pressure for the upper and lower wings are vertically, or closely, in line. Aerodynamically, this is not a suitable arrangement, hence most biplanes employ stagger.



**Photo:** Most biplanes have either positive or negative stagger. This photo shows a Douglas O-38F with forward stagger, on display at the National Museum of the US Air Force in Dayton, Ohio.

Stagger reduces biplane interference and generally improves aerodynamic efficiency and increased longitudinal stability, particularly with positive stagger. It also provides better pilot visibility, maximum lift and with forward centre of pressure movement at increased angles of attack, the top wing tends to stall before the lower wing, leading to a more docile stall.

True stagger or effective stagger is used for calculations in aerodynamic research and is measured at zero degrees angle of

attack. The first aircraft to use staggered wings was the Goupy II, built in 1909 by the Bleriot factory in France.

*See* **Biplane interference; Orthogonal biplane.**

**Stagnation entropy** *See* **Entropy.**

**Stagnation point/line** The stagnation point/line is located on the airfoil's leading edge surface where the approaching airflow impacts it and becomes stationary.

The airflow approaching the wing's leading edge possesses ambient static pressure and dynamic pressure. At the leading edge, the airflow velocity reduces to zero (it stagnates). Dynamic pressure is converted to increased static pressure, which is equal to the total pressure – ambient pressure plus dynamic pressure, in accordance with Bernoulli's principle. The stagnation point/line static pressure will be higher than the ambient pressure by an amount equal to the free-stream dynamic pressure.

With increasing angle of attack, the stagnation point/line moves lower on the leading edge, in effect increasing the upper surface wing area, creating increased lower pressure and hence more lift. The reverse effect occurs on the lower surface, where the under surface area is decreased, resulting in less change in air pressure. Therefore, the increased air pressure differential between the upper and lower surfaces results in an increase in lift with increasing angle of attack. However, if the wing has a small leading edge radius, a very low-pressure area can form above the stagnation point/line contributing to the already present adverse pressure gradient. The stagnation point/line plays a very important part in the production of lift!

A rear stagnation point also exists at the wing's trailing in association with the Kutta condition.

The term stagnation point is used in reference to a two-dimensional airfoil section as opposed to the stagnation line, which is used in reference to a three-dimensional airfoil.

*See* **Stagnation pressure; Forward stagnation point.**

**Stagnation pressure** The air pressure brought to a rest on the wing's leading edge is known as the stagnation pressure, represented by the symbol  $p_0$ .

Stagnation pressure is the sum of the dynamic pressure ( $\frac{1}{2} \rho V^2$ ) plus the local static pressure, the same as recorded by a Pitot tube. The pressure value depends on aircraft speed and altitude. The aircraft's nose and wing leading edges experience impact (stagnation) air pressure at their stagnation points, which increase in pressure when brought to rest. Above and below the stagnation point at the wing's leading edge, the air pressure is decreased and increased as it flows over and below the wing respectively. Stagnation pressure and local static air pressure are related by the free stream Mach number.

See **Static air pressure; Total pressure.**

**Stall** Wings cease to produce lift at high angles of attack (usually around 12 to 15 degrees) at the point of the maximum lift coefficient, where the attached airflow over the wing fails to follow the upper surface curvature, becomes turbulent, and breaks away. Stall speed is always recorded as indicated air speed and remains the same at all altitudes, with true air speed increasing with altitude. An aircraft can stall at any speed greater than its basic stall speed and in any attitude – during a turn, a climb, or descent, straight and level, inverted and with an increase in weight. However, the critical angle of attack is always the same. Flaps and other high lift devices can lower stall speed and/or increase the stall angle.

Several factors determine the stalling speed and the stall's characteristics:

- shape of the wing section
- thickness/chord ratio and position of maximum thickness
- planform and aspect ratio
- wash-out
- slats and slots
- flaps

- power
- wing loading.

The effect of the maximum lift coefficient, wing loading, altitude (density ratio) and wing area on stall speed is shown by the following formula:

$$V_s = 17.2 \sqrt{\text{weight}/C_L \text{ max. } \sigma \cdot S}$$

Where  $V_s$  = stall speed KTAS

$W$  = gross weight

$C_L \text{ max}$  = maximum lift coefficient

$\sigma$  = density ratio

$S$  = wing area.

The stall speed is normally quoted as indicated air speed (IAS); however if the speed is required in equivalent air speed (EAS) then the density ratio of 1.000 for sea level is used.

In 1904, Wilbur Wright introduced the term ‘stall’.

**See Self-induced stall; Positive stall; Progressive stall; Power-on stall.**

**Stall break** The stall break occurs when the critical angle is reached and the nose pitches down relative to the aircraft’s OY lateral axis.

When the angle of attack increases, the centre of pressure moves steadily forward until the wing stalls. At that point, the airflow over the wing becomes turbulent, breaks away and the centre of pressure quickly moves rearwards on the wing. The increased moment couple between the centre of gravity and centre of pressure causes the nose to pitch down – the stall break.

Several factors determine the characteristics of the stall break. Undesirable stall behaviour can be avoided by designing features into the wing to ensure a stable stall break, such as taper, washout, stall strips or leading edge devices, etc. Sweptback wings are more

prone to an unstable stall break and pitch upwards, as opposed to a straight wing, which normally pitches downwards.

The characteristics of the stall break can be shown on a diagram for lift coefficient versus angle of attack. A gently rounded peak (a rooftop curve) on the lift coefficient line at the critical angle indicates a gentle stall usually associated with a supercritical wing. Conversely, a small radius, peaky curve at the critical angle on the lift coefficient line indicates a more active stall break.

*See **Diagram 49, Lift Coefficient V. Angle of Attack; Slats; Slots; Krueger flaps.***

**Stall fence** A stall fences is used primarily on sweptback wings to prevent a spanwise flow of air, which can lead to a wingtip stall condition.

*See **Fence.***

**Stalling angle** The stalling angle is the angle of attack at which the lift coefficient reaches its maximum value, the main wings stall with a loss of lift, the centre of pressure moves aft and the nose pitches down on aircraft with straight wings; sweptback wing aircraft may pitch nose-up.

The term stalling angle, although in common use, should correctly be replaced with critical angle of attack.

*See **Diagram 49, Lift Coefficient V. Angle of Attack; Critical angle of attack; Lift coefficient; Maximum lift coefficient.***

**Stall progression** A wing does not normally stall over all its surface area at once, but progresses from one particular area and spreads across the remainder of the wing. Where the stall initially develops depends on the wing planform; it is always preferable for the stall to progress from the wing root area and spread towards the tips. This produces a square stall and ensures the ailerons remain effective into the stall. A wingtip stall is undesirable due



to the adverse effect on ailerons and a possible asymmetric stall, which could develop into a spin.

Compared to other wing planforms, rectangular wings as found on the early Piper PA 28 Cherokee light aircraft have a relatively lower lift coefficient. This is due to the high-pressure air leaking around the wingtips and unloading the wing, with the advantage the wing root stalls before the wingtips.

Elliptical, sweptback, and tapered wings have relatively higher lift coefficients and tend to stall at the wingtips first. The stall then progresses inboard towards the wing roots; stall fences are common on these types of wings to straighten out the airflow to induce a chordwise flow across the wing. Stall strips are mounted on some wings to induce root stalling. It is preferable for the wing tips to stall last and recover first to maintain control.

*See* **Stall**.

**Stall region** The stall region is located at the helicopter's main rotor blade root area. [In normal powered flight, stalling may occur on the retreating blade tip area, but it is the blade root area that is stalled during autorotation]. Due to the relatively high rate of autorotation descent, the relative airflow enters the rotor disc from below, which results in a large angle of attack at the blade root. The lift vector is tilted well forwards of the axis of rotation (it is always at right angles to the relative airflow). The rotor drag is high due to the high angle of attack causing the total reaction to lean well back, which causes the rotor RPM to be retarded. The lift/drag ratio is not very efficient in the blade root region.

*See* **Autorotation – helicopter**.

**Stall speed** The wing can stall at virtually any air speed if the critical angle of attack is reached and exceeded; the angle of attack is the determinant of stalling, not speed. However, pilots use various air speeds for reference as to when the wing will stall, rather than angle of attack. Stall speed varies with flap position, aircraft weight, power on or off and wing loading. Stall speed on a piston-engine aircraft is slightly lower with power on compared to power off due to the propwash over part of the wing and prop thrust being inclined upwards. The stall speed is also in proportion to the square of the aircraft weight and inversely proportional to the wing area and the maximum lift.

In a glide, the basic stall speed will be a few Knots lower than in straight and level flight in the same configuration. This can be explained by the lift being equal to the weight times cosine the glide angle; the lift in a glide is less than the weight and the stall speed is therefore, lower. Conversely, an increase in wing loading such as during a steep turn or other positive flight manoeuvres will raise the stalling speed. The indicated stall speed is recorded in the Pilot's Notes based on the centre of gravity being at the forward limit, where the stall speed is higher than when the CG is at an aft position.

The following formula shows the increase in stall speed for a given bank angle:

$$V_{so \phi} = V_{so} \sqrt{n}$$

Where  $V_{so \phi}$  = stall speed at selected bank angle

$V_{so}$  = stall speed, flaps up

$n$  = bank angle load factor.

From the above formula, the percentage increase in stalling speed is shown in the following table:

| Angle of Bank | Percentage Increase in Stall Speed |
|---------------|------------------------------------|
| 30°           | 7%                                 |
| 40°           | 14%                                |
| 50°           | 25%                                |
| 60°           | 41%                                |
| 70°           | 71%                                |
| 75°           | 100%                               |

A change in aircraft weight has an effect on the stall speed. Light aircraft have little change in stall speed because the allowable weight change is not very large. However, large aircraft vary their weight quite considerably with fuel burn on long trips and varying payloads and the change in stall speed can amount to several Knots. An increase in stall speed of 1% can be the result of a 2% increase in weight, or vice versa.

The formula below shows the change in stall speed due to weight, with density ratio (altitude), wing area, and maximum lift coefficient remaining constant:

$$V_{s^2}/V_{s^1} = \sqrt{W^2/W^1}$$

Where  $V_{s^1}$  = stall speed at a given weight ( $W^1$ )

$V_{s^2}$  = stall speed at second weight ( $W^2$ )

**See Stall; Turning.**

**Stall-spin** When an aircraft wing stalls with an associated yaw, the aircraft will enter into an incipient spin, with a full spin developing if not brought under control immediately. Rudder should be used to control direction during the incipient stage, not ailerons. The direction of spin is determined by the slower wing stalling first and dropping to initiate the spin.

**Stall strip** A stall strip is a stall promoter. It consists of a metal strip of short length with a sharply defined forward edge, placed on the inboard leading edge of the main wings. Its purpose is to trip the airflow near the wing roots into a stall condition before the remainder of the wing stalls. The stall strip effectively makes the leading edge radius smaller, which tend to stall at lower angles of attack compared to a similar wing with a larger leading edge radius. The ailerons may still maintain roll control without aggravating the incipient stall stage loss of control. Note: it is normal practice not to use ailerons during stall recovery due to their aggravating affect on roll control on some aircraft! Use rudder only to maintain direction.

The stall strip can raise the stalling speed a couple of Knots inducing the stall to occur earlier over the wing root area than it would otherwise. However, the stall speed increase would be marginal on most light aircraft; improving stall behavior takes priority for safety reasons.

Also known as a spoiler, flow strip, or anti-spin strip.

**Stall/surge** A stall/surge refers to the breakdown of the smooth, orderly airflow through a gas turbine (jet) engine. Depending on its severity, it can be heard as a low rumble or a louder explosive sound. It can induce the following:

- loss of thrust
- RPM variation
- fuel flow variation
- rapid increase in turbine temperature
- flameout (engine failure) in some situations.

**Stall warning vane** The stall warning vane is located on the main wing's leading edge stagnation line and is there to activate the stall warning horn/light in the cockpit. At normal angles of attack (0 to 15°), the airflow holds the spring-loaded stall vane switch closed. An increase in angle of attack close to the critical angle will change the location of the stagnation line from above the vane to below it, diverting airflow from a lower angle, which will lift the vane and activate the stall warning system before the stall occurs. The stall warning should activate at least 10 Knots before the stall occurs.

The stall warning vane system (also known as a stall or lift detector) was invented by Leonard Greene of Safe Flight Instruments Co, in 1944. It is the main system used on thousands of light aircraft; heavier aircraft use alternate systems.

**Standard atmospheric pressure** Air pressure is considered to be at its standard pressure value when it supports a column of mercury 0.76 meters (29.92 inches) high at zero degrees Centigrade at sea level. Standard pressure units in common use in aviation are 14.7 PSI or 1013.25 hPa.

*See* **Pressure.**

**Standard density** In meteorology, the standard density is assumed by the International Standard Atmosphere to be 1.225 kg/m<sup>3</sup> or 0.516 g/cm<sup>3</sup>.

*See* **Absolute density; Density ratio; Standard density; Relative density.**

**Standard mean chord** The standard mean chord is defined by the wing's gross area divided by the span (gross area/span).

Also known as the geometric mean chord.

**Standard pitch** The propeller blade's geometric pitch is by convention, measured at approximately the 66% station or two-thirds the distance from the propeller hub to the tip. This location is known as the standard pitch.

It is also known as the nominal pitch.

See **Standard radius**.

**Standard radius** The standard radius of a propeller is located at about the 66% station. On the propeller hub, a set of numbers such as 80" × 57" can be found. The first number refers to the diameter of the propeller in inches, while the second number refers to the geometric pitch in inches at the standard radius.

See **Geometric Pitch; Diagram 68, Propeller Pitch**.

**Standard rate turn** A standard rate turn takes the aircraft two minutes to turn through a full 360 degrees at a rate of 3°/second.

It is also known as a two-minute turn or a Rate 1 turn.

**Starting vortex** At the commencement of the aircraft's take-off roll, lower air pressure develops on the wing's upper surface in accordance with Bernoulli's theorem. The airflow leaks from underneath the wing around the trailing edge as a fast moving flow. Two stagnation points develop, one on the leading edge and an aft stagnation point on the wing's upper surface just forward of the trailing edge. A starting vortex forms between the aft stagnation point and the trailing edge as the circulation develops to become a trailing edge (or sheet) vortex. A bound vortex (bound to the lifting line) forms along the length of the wing between each tip.

The starting and bound vortices rotate in opposite directions of equal intensity being two equal halves of the closed loop circulation; they gradually increase in strength to form the established airflow over and under the wing to produce lift. The aft stagnation point settles at the trailing edge. The starting vortex dissipates and is shed

from the wing into the trailing wake. Wing tip vortices develop as part of the closed loop's starting vortex and circulation, which remain for the duration of the flight. The tip vortices become of significance to following aircraft (wake turbulence) beginning at the point of rotation. When the initial airflow has stabilized at the commencement of take-off, circulation is continuous and the air flows off the trailing edge with a downward component in accordance with the Kutta condition. All the above happens quickly and circulation is well established by the time the wing has moved forward about two chord lengths.

Professor Ludwig Prandtl (1880–1953) discovered the starting vortex during a wind tunnel experiment on the circulation around a wing.

**See Rear stagnation point; Kutta condition; Kelvin's circulation theorem.**

**Static air pressure** In the aviation industry frequent reference is made to fuel, oil, hydraulic and air pressure. Pressure, the force per unit area, can be defined as static or dynamic pressure.

The physicist would describe the static air pressure by using the molecule theory; every fluid including air is made up of an infinite number of molecules. When air for example is at rest, the molecules exert a uniform pressure in all directions due to their random motion. This uniform molecule pressure is known as static pressure.

The symbol ' $p_o$ ' represents the sea level static air pressure, and the ambient static pressure is represented by the symbol ' $p$ '. The air pressure reduces with an increase in altitude and at 18000 feet; it is at half of the sea level pressure.

Our bodies feel the static pressure of air. Some may argue they cannot feel it but air pressure at sea level is acting on our bodies (and everything else) with a force of approximately:

- 101.3 kN/m<sup>2</sup>
- 14.7 lbs./sq. inch.

- 2116 lbs./sq. foot
- 29.92 inches Hg.

Our body produces an equal and opposite reaction or outward force, that is why we do not feel the air pressure. If we are outside on a windy day, we can feel the force of the wind pushing on our bodies. The molecules of air are now traveling in the direction of the wind and we feel this as dynamic pressure. The same applies to the aircraft. When the aircraft is stationary on the ground and assuming no wind, only the static air pressure is acting on the aircraft.

Also known as the free-stream static pressure.

*Compare with Dynamic pressure. See Stagnation pressure; Total pressure; Altitude pressure ratio.*

**Static balance** The term static balance can be applied to the aircraft as a whole, or to its control surfaces. The aircraft is in a state of static balance when the moments about all its axes are zero.

Static stability exists on the control surfaces when there is zero torque force. Either a mass balance or the placement of the control surface's centre of gravity on the hinge axis balances the control surfaces.

**Static instability** Static instability is an undesirable stability characteristic and may be due to an improperly loaded aircraft or one of faulty design.

When an aircraft is upset in flight (generally in the pitch attitude from straight and level flight), the aircraft's flight path diverges away from the required attitude. Static instability is not acceptable under any circumstance.

**Static lateral stability** *See Lateral static stability.*



**Static margin** The static margin is associated with an aircraft's static stability characteristics. It represents the margin of static stability and is the distance measured from the neutral point to the centre of gravity location, expressed as percentage MAC.

At a forward centre of gravity the static margin and hence, the stability is greater but as the centre of gravity moves aft decreasing the static margin, static stability also decreases. Therefore, stability is proportional to the static margin. The static margin is a variable distance as opposed to the minimum safety margin, which is a fixed distance.

Sidney B. Gates (1893–1973) introduced the static margin to aerodynamics while working at the Royal Aircraft Establishment at Farnborough, UK.

*See Diagram 12, Centre of Gravity Range; Diagram 63, Pounds of Pull V. CG Percentage MAC; Diagram 78, Static Stability V. Angle of Attack.*

**Static port** The static port is part of the Pitot-static system, which provides static and dynamic air pressure to the ASI, VSI and altimeter. The static port, which feeds free stream static air pressure to the system, may or may not be combined with the Pitot tube. An alternate location is to place a flush mounted static port on each side of the fuselage to account for any disruption in the smooth airflow into the system, which may cause position error due to the yaw or pitching of the fuselage. The location of the static port on the aircraft is important due to it creating greater position error than that caused by the Pitot tube. Some light aircraft may have a single port located in the tail cone.

*See Pitot-static system.*

**Static propeller thrust** The static propeller thrust is the thrust produced at maximum RPM when the aircraft is stationary.

The static propeller thrust can be calculated from the following formula:

$$\text{Static propeller thrust} = \text{PDA} \times V_1 \times \rho \times V_o$$

Where PDA = propeller disc area, sq. meter or (sq. ft.)

$V_1$  = inflow velocity, meters per sec or (FPS)

$\rho$  = air density  $\text{kg/m}^3$  or  $\text{lbs/cu.ft.}$

$V_o$  = outflow velocity, meters per sec or (FPS).

Note, when using Imperial units the answer will be in pounds of thrust and for SI units the answer is in kilogram force. The difference in the formulas between static propellers thrust and cruise thrust is the addition of the aircraft velocity ( $V$ ) in the cruise thrust formula.

*Compare with* **Cruise thrust.**

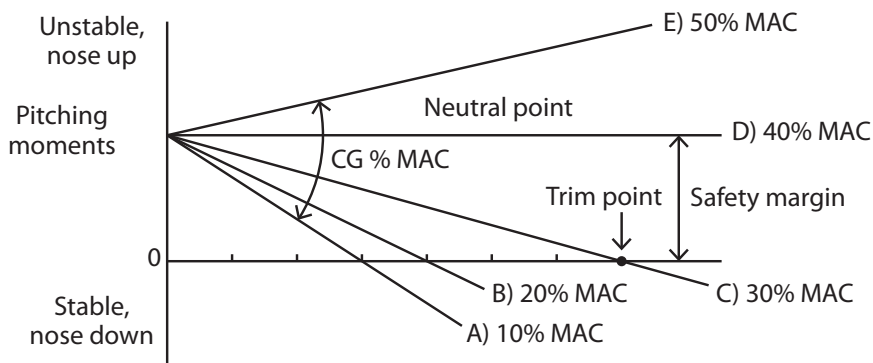
**Static stability** Static stability is a measure of the aircraft's initial *tendency* to return or diverge from its trim speed after an in-flight upset. On the other hand, dynamic stability is the *nature* of the aircraft's motions after an in-flight upset.

Positive static stability is achieved only when the aircraft is correctly loaded and with the centre of gravity within its specified limits. This is the only acceptable stability condition for safe flight.

Associated with positive static stability is dynamic stability, which is acceptable only if that also is positive. If the aircraft exhibits neutral dynamic stability, the oscillations will continue at a constant amplitude without any tendency to damp out or to diverge further. Negative dynamic stability results in the oscillations increasing in magnitude and diverging further from the trim speed.

If the aircraft is disturbed from its trim speed, a restoring force sets in which results in corrective motion to regain the trim condition. Static stability initially produces a short-term oscillation

in pitch, which is followed by a long-term phugoid motion. If the dynamic stability is positive, it will damp out the oscillation to a minimum.



**Diagram 78, Static Stability V. Angle of Attack**

Reference to *Diagram 78, Static Stability V. Angle of Attack*, shows the effect the centre of gravity location has on an aircraft's static stability. The degree of stability is indicated by the angle of the slope on *Diagram 78*. Curve 'A' for 10% MAC has the steepest slope proving stability is greater with a forward centre of gravity. Stability gradually reduces as the centre of gravity moves rearwards towards the aft centre of gravity position; in effect, the line moves up the page, up to curve 'C' for 30% MAC (assume 30% MAC is the aft centre of gravity limit). In its most acceptable condition the aircraft will, after a disturbance from its flight path, return to straight and level flight with a short term, dead beat oscillation. This is the most ideal of all stability conditions. Nevertheless, positive stability is more common with a damped oscillation; the aircraft experiences several oscillations of decreasing magnitude.

Conventional aircraft require positive static stability to ensure it has sufficient, safe stability and adequate control. However, some fighter aircraft are intentionally designed with built-in reduced, or even negative static stability to increase their manoeuvrability

during combat. These aircraft require an autopilot with computer assist to provide normal control feel for the pilot. Apart from greater manoeuvrability, a smaller stabilizer can be installed reducing cruise trim drag. The General Dynamics F-16 Fighting Falcon was an early fighter with built-in negative static stability.

Curve 'D' at 40% MAC represents neutral stability, or neutral equilibrium. When the centre of gravity is located at the stick-free neutral point, neutral static stability will exist. If the aircraft is disturbed from its attitude, it will tend to stay in the new position without any attempt to return to its original attitude. Controlling the aircraft will be difficult.

With the centre of gravity well aft of the stick-free neutral point at 50% MAC (line 'E') the aircraft is in a dangerously, unstable negative static stability condition. In this condition, an offset from its flight path will cause the aircraft to diverge further and further from the required attitude. The aircraft will be extremely difficult to fly.

Note, on **Diagram 78**, a negative down-slope (curves A, B, or C) represents positive static stability; the horizontal line (curve D) represents neutral stability and negative stability is represented by the up-slope (curve E).

*See Lateral static stability; Lateral dynamic stability; Dynamic stability; Short-term (or short-period) oscillation; Phugoid motion; Positive static stability; Stick-free; Stick-fixed; Diagram 12, Centre of Gravity Range. Compare with Diagram 61, Pitching Moments V. Angle of Attack.*

**Static strength** The limit and ultimate load factors determine the maximum safe static strength for the aircraft.

**Static thrust** Static thrust is produced by a jet engine when it is stationary. The air is sucked in through the intake, heated, and ejected out the tailpipe with a much higher velocity than at the intake. This difference in the velocity of the airflow produces the static thrust. Static thrust is measured when the aircraft is stationary (no ram air), in standard ISA conditions at sea level. It is used mainly to compare the power of different jet engines and it can be shown by the following formula:

$$F = M(V_2 - V_1)$$

Where F = force (thrust)

M = mass of airflow

$V_2$  = exhaust velocity

$V_1$  = 0 (forward speed and inlet velocity  
is zero when stationary).

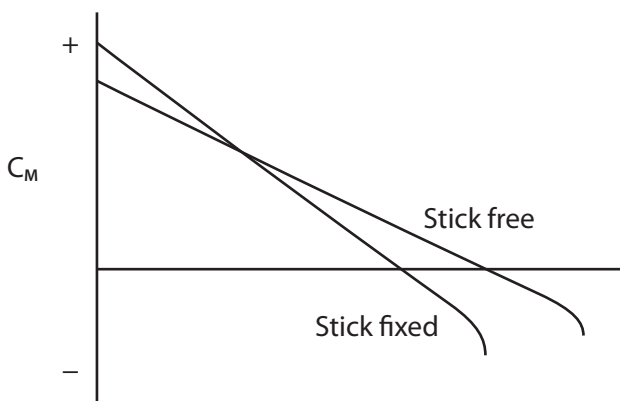
Also known as the rated static thrust or gross thrust.

**Station** The station is a given location on the aircraft's OX longitudinal axis, which is identified by a number equivalent to the distance in inches (or millimeters) from the reference datum. The reference datum – normally the forward face of the firewall or some other arbitrary point near the front of the aircraft, is labeled as station zero. The terms station and moment arm are used interchangeably. Stations are used mainly for weight and balance calculations by flight crew.

See **Moment arm**.

**Station zero** See **Reference Datum**.

**Stick-fixed** When the elevators are held in the trimmed position by the pilot it is said to be stick-fixed. Stick-fixed (or stick-free) both have an effect on the degree of static longitudinal stability. *Diagram 79, Stick-fixed and Stick-free*, shows the stick-free slope is at a lesser angle than the stick-fixed slope indicating stability is less for the stick-free condition.



**Diagram 79, Stick-fixed and Stick-free**

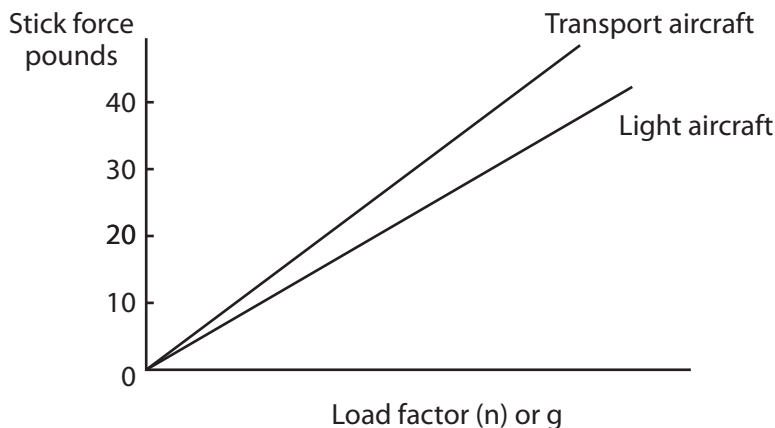
Most helicopters are statically stable but dynamically unstable in the longitudinal plane when flying stick-fixed.

It is also known as, stick held.

Compare with **Stick-free**.

**Stick force gradient** The term stick force gradient describes the graph slope showing the stick force in pounds of pull versus limit load factor. This graph shows the relative differences in the gradients for fighter and transport aircraft; the gradient line for fighter aircraft is located below the gradient line for transport aircraft. Note the stick force gradient increases with an increase in load factor and decreases with increasing altitude.

It is also known as the manoeuvring stick force gradient.



**Diagram 80, Stick Force Gradient**

**Stick force per 'g'** Stick force per 'g' per Knot lost is an alternate name for pounds of pull.

It is the force required to raise the aircraft's nose, or in other words, the pounds of pull required to move the control column rearwards. Stick force per 'g' becomes greater increasing linearly with higher speed, greater control deflection and with forward centre of gravity. Greater stick force means the elevator becomes harder to move requiring more muscle force by the pilot.

Light, sport aerobatic aircraft and jet fighters have very light control forces (3–8 lbs./'g' is common) making them highly

manoeuvrable as opposed to transport aircraft, which have heavier stick force gradient with less manoeuvrability. An aircraft with a light stick force gradient could more easily be overstressed; therefore, it is built with a higher limit load factor to withstand the extra stress forces involved with manoeuvring.

While working at the Royal Aircraft Establishment at Farnborough, UK, Sidney B. Gates (1893–1973) introduced the term stick force per ‘g’ to aerodynamics.

*See Pounds of pull; Diagram 63, Pounds of Pull V. CG Percentage MAC; Diagram 80, Stick Force Gradient.*

**Stick-free** When the tailplane/elevator is in the trimmed position and released by the pilot to float freely, it is said to be stick-free. In this condition, the tailplane produces less stability compared to a stick-fixed elevator due to less lift on the tailplane.

Most helicopters are statically unstable and therefore have little or no dynamic stability in the longitudinal plane. Hence, the greater difficulty to fly a helicopter compared to an airplane.

*See Diagram 79, Stick-fixed and Stick-free.*

**Stick held** *See Stick-fixed.*

**Stick position stability** By convention, pulling the control stick back raises the aircraft’s nose, which increases the angle of attack, and pushing forward lowers the nose to decrease the angle.

Positive stick position stability is only achieved when the centre of gravity is located forward of the stick-fixed neutral point.

**Stokes, G.** The Englishman, George Gabriel Stokes (1819–1903) independently developed the Navier-Stokes equation with Claude-Louis Navier (1785–1836). Both men are credited with establishing the now famous scalar equation during the first half of the 19<sup>th</sup> Century. The equation was commonly used and still



is, by aerodynamicists and scientists for aerodynamic calculations during their research, for example into three dimensional, compressible, viscous flows.

See **Navier-Stokes Equations**.

**STOL** STOL is the acronym for short take-off and landing.

A STOL aircraft is one with high-lift devices designed to produce maximum lift coefficient at relatively low air speeds. Large flaps, slots or slats, leading edge droop, large wing area and high-lift cambered wings are all common on STOL aircraft. Usually V/STOL (vertical and short take-off or landing) is very closely associated with the STOL acronym.



**Photo:** The Scottish Aviation Twin Pioneer is a high-wing transport with remarkable short field STOL capability. The Twin Pioneer is on view at RAF Museum Cosford, UK.

**Straight stall** A straight stall is one where both main wings stall simultaneously, following the use of rudder to prevent any adverse yaw. It is the ideal stall condition.

Also known as a square stall.

**Straight wing** A wing that is neither of sweepback or forward sweep, nor of delta configuration is classified as a straight wing. A wing with a straight line joining both wingtips and passing through the wing roots qualifies under this definition. A few degrees of leading edge sweepback or trailing edge sweep forward or an elliptical wing are acceptable under the definition of a straight wing.

**Strakes** A low aspect ratio wing leading edge root extension is mounted on the forward fuselage ahead of and a part of the main wing. Its purpose is to induce a high-speed vortex, which induces vortex lift.

With strakes attached, a smaller, lower aspect ratio wing can be used saving weight. Higher angle of attack manoeuvres above the normal critical angle can be achieved with the added advantage of greater directional stability and higher roll rates, an important requirement for a manoeuvrable jet fighter.

The Concorde has a strake that is beautifully merged in with its delta type wing. However, strakes are more common on high-speed jet fighter type aircraft. The Northrop Grumman X-29 research aircraft was unusual in having strakes running along the side-rear fuselage with small flaps on their ends.

Strakes are also mounted on the left-hand side of a helicopter's tail boom. It is used for anti-torque control by spoiling the main rotor downwash as it flows over the tail boom. The downwash over the right-hand side of the tail boom flows freely and smoothly and is encouraged to stay attached due to the Coanda effect. Less work is required from the tail rotor and so smaller blade pitch angles and therefore less engine power is required. The greatest effect is found in the hover but it reduces with increasing forward speed.

However, increasing directional stability with increasing speed overcomes the reduced strake effect.

*See* **Forebody strake; Aft body strake; Chine; Vortex lift.**

**Streamline** A streamline is a line on a diagram showing the directional path of small parcels of air particles (fluid elements) in a steady state laminar flow. The streamlines become more compressed as the pressure decreases, typically over the upper surface of the wing, showing an increase in airflow velocity. It follows, if the streamlines are further apart, they represent a decreased velocity. Bernoulli's theorem is used to relate the velocity and pressure along a streamline.

*See* **Bernoulli's theorem.**

**Streamline shape** A smoothly contoured, 3-D shaped body as found on a modern aircraft, is streamlined to reduce the effects of air resistance, or drag. It is the opposite of a bluff body, which causes increased aerodynamic drag.

*See* **Drag.**

**Streamline flow** A laminar airflow over a smooth surface produces a streamline flow. The streamline flow is of a constant direction or slightly curved and devoid of any irregularities. It is found in a perfect fluid and one without viscosity.

*See* **Flow line.**

**Streamwise vortices** *See* **Wingtip vortices.**

**Structural Stress** The structural stress acting on an aircraft due to an applied load can be divided into four main categories of torsional, tensile, compression, and bending stress. Any one stress or combinations of stress can be present acting on an aircraft or

propeller at any one time. Stress is measured as a force per unit area. All aircraft are subject to stress and must be flown within the operating envelope to ensure the stress limitations are not exceeded. This applies to the airframe, engine, and propeller.

*See* **Propeller stress; Manoeuvre envelope.**

**Structure & composition of the atmosphere** The air mass immediately surrounding the Earth consists firstly of the troposphere with the stratosphere above with the two regions separated by the tropopause. The troposphere, which contains 80% of the atmosphere, is our main interest here.

The composition of the atmosphere remains unchanged up to a great height but the change in physical conditions does vary. These conditions are:

- A lapse rate of  $1.98^{\circ}\text{C}$  up to the tropopause, which is at a height of approximately 55000 feet over the equator, lowering to around 25000 feet over the North and South poles.
- The tropopause varies in height on a daily basis depending on the temperature with a distinctive overlap of the tropical and polar tropopause around the  $35^{\circ}$ – $50^{\circ}$  latitudes north and south.
- The majority of the weather is confined to the troposphere as opposed to the stratosphere, which is relatively clear and stable due to the temperature being almost constant.

Air is not a gas in its own right but a mixture of different gases of varying amounts as set out in the table below. Nitrogen, oxygen, and argon are the three main gases and the remainder is trace gases, which make up the remaining 0.039% of the dry atmosphere. Water vapor is also present in varying amounts from about four percent down to almost zero percent. Antoni Marti Franques (1750–1832) was the first person to calculate correctly the composition of air circa 1784.

| Gas                                                     | Volume %    | Weight % |
|---------------------------------------------------------|-------------|----------|
| Nitrogen                                                | 78.09       | 80.5     |
| Oxygen                                                  | 20.95       | 23.1     |
| Argon                                                   | 0.93        | 1.3      |
| Carbon Dioxide                                          | 0.03        | 0.05     |
| Hydrogen, Neon, Helium, Ozone,<br>Radon, Krypton, Xenon | Traces only |          |

*See Atmosphere; International Standard Atmosphere.*

**Struts, landing & flying wires** Wing struts, or interplane struts, are used in tandem on biplanes (and triplanes) to form a rigid construction between the upper and lower wings. They are braced together either by a solid third strut placed diagonally between the two vertical struts to form an N-strut, or by diagonal incidence wires. Flying wires are attached diagonally from the lower wing to a point further outboard on the upper wing. Landing (or bracing) wires run from the lower wing to appoint further inboard on the top wing; they are also known as anti-lift wires. The flying and landing wires are ground-adjustable for rigging the aircraft's wings to the required dihedral and angle of incidence.

Bracing struts are used on a few monoplanes, for example, the Cessna fleet of high wing, single-engine aircraft, which use a single strut to each wing. Early monoplanes mostly have a double or V-strut, where each strut attaches to the fuselage at the same place and diverges out to attach to the wings under surface. The disturbed lift on the under surface due to flow interference between the wing and strut, is of less concern than disturbed flow on the upper surface, which provides the majority of the wing's total lift. This is the reason why struts are less popular on low wing aircraft; the Cessna 188 Agwagon is an exception.



**Photo:** The Fairchild Argus has solid V-struts, which also help support the undercarriage. This Argus is on display at RAF Museum Cosford, UK.

In flight, the bracing struts are referred to as ties, not lift struts; however, the term struts applies on the ground. They perform the same function in flight, as do a biplane's flying wires – they are in tension, the wing supports the fuselage. On the ground, the bracing struts are akin to the biplane's landing wires – they are under compression and support the wing against gravity.

On low wing monoplanes, the wing struts in flight are under compression, and on the ground, they are in tension – the opposite of high wing aircraft. Wire bracing may be used in lieu of struts due to their reduced drag and weight.

A cantilevered wing is one devoid of any struts or external wire bracing; all supporting structure is contained within the wing to reduce drag.



**Photo:** The Scottish Aviation Twin Pioneer has a stub wing for bracing the main undercarriage. The RAF Museum Cosford, UK is home to this Twin Pioneer.

**Stub wing** A stub wing is a very short span, tapered wing, which is normally used to structurally brace the main undercarriage assembly to the fuselage. It is too short a span for the plane to be classed as a sesquiplane, hence the term stub wing.

**Subsonic** Subsonic is considered the speed of a moving object (aircraft) below Mach 1.0, the speed of sound, although the transonic speed overlaps the subsonic speed from Mach 0.8 to Mach 1.2, in the supersonic region.

**Subsonic flow** The speed of a subsonic airflow is less than Mach 1.0, the speed of sound. A subsonic flow acts very much like flowing water. It obeys Bernoulli's principle where a static pressure increase is accompanied by a decrease in velocity and vice versa, with a negligible variation in air density.

Subsonic flow can be further divided into low-speed and high-speed flows. The low subsonic flow is an incompressible flow between 0 and 200 Knots (Mach 0.3). Above this speed, changes in air density must be taken into consideration and the flow is considered compressible.

**Suction face** An operating propeller induces air to flow in through the front of the propeller disc (the suction face) causing a low-pressure area and exit the rear of the disc causing a high-pressure area.

*See Cavitation.*

**Suction surface** The term suction surface is an alternate name for the upper surface of an airfoil. The suction or lower pressure is induced by an increase in flow velocity over the wing. The suction surface pressure is at a lower value than the increased pressure below the wing, which is known as the pressure surface.



**Superaerodynamics** Superaerodynamics involves speeds in the supersonic region and low air density. The speed is such that the pressure waves projected ahead of the aircraft have no time available for the air molecules to collide with each other and warn the air upstream of the approaching aircraft.

It is also known as Newtonian aerodynamics; the newtonian sine-squared law applies to high-speed air flows.

*See* **Supersonic airflow; Free-molecular flow.**

**Supercirculation** Supercirculation is a form of boundary layer control by using engine power to blow extra air over the wing's upper surface, or to suck the boundary layer into the wing, to enhance the circulation.

*See* **Boundary layer control system.**

**Supercritical airfoil** A supercritical airfoil is one that reduces drag in the high subsonic speed region where the airfoil may encounter supersonic flow over parts of the wing. The speed range between and including the critical Mach number (at about Mach 0.8) and the drag divergence Mach number (at about Mach 0.9) is increased. These types of airfoils are becoming increasingly popular with business jets and jet transport aircraft.

The airfoil shape generally has a large radius leading edge with a reduced camber on the upper surface, which reduces the lift produced over the forward part of the wing. The wing is described as a roof top section, or aft loaded wing with increased suction area located over the aft 30 % of the wing; the wing is rear-loaded due to a more prominently bulged, forward, lower surface with a cusp, or reflex camber on the under surface at the trailing edge. Greater lift is produced on the forward, under surface increasing lift and a positive pitching moment, compared to a normal wing. Less drag is the result of an improved and stabilized flow.

Other advantages of a supercritical wing are:

- a progressive stall break
- boundary layer shock induced separation is reduced
- shock waves develop further aft than normal location and at a higher speed
- a higher drag divergence Mach number due to a relatively low thickness ratio with reduced upper surface camber
- the greater depth of the wing chord allows for greater fuel volume and requires less sweepback angle for a given Mach cruise speed
- Wave drag is reduced.

The disadvantage of supercritical airfoils is:

- airflow separation on the aft, lower surface can be caused by an adverse pressure gradient.
- too much aft loading of the airfoil can cause negative pitching moments.
- Operation at off-design limits can increase drag with increasing Mach number and adversely affect the formation of lift.

Richard T. Whitcomb (1921–2009) an aerodynamic researcher at NACA Langley Research Centre invented the supercritical airfoil in 1965. He was also responsible for the invention of area rule; both were important inventions and great assets to aircraft designed to fly in the high subsonic region.

A supercritical wing was first flight tested on 24 November 1969 on a modified North American Rockwell T-2 Buckeye and later, further testing was performed using an LTV Vought F-8A naval jet fighter on 31 March 1973.

*See Cusp; Critical Mach number; Sweepback angle ( $\Lambda$ ); Diagram 21, Drag Divergence Curve.*





**Photo:** A Vought F-8 Crusader similar to the plane used to flight-test a supercritical wing. This example is located on the USS Midway Museum's aircraft carrier in San Diego, Ca.

**Supercruise** Supercruise is the term given to an aircraft capable of sustained flight at supersonic speed without the need for afterburner assistance, while carrying the designed payload. Several aircraft are capable of supercruise including Concorde and the English Electric Lightning, which was the first aircraft in the world to achieve supercruise ability, on 11 August 1954.

The term was coined by Fighting Falcon designers, USAF Col. John Boyd, Col. Everest Riccioni, and Pierre Sprey.

**Super-manoeuverability** An aircraft with greater than normal manoeuvrability at high angles of attack beyond the normal stall is said to have super-manoeuverability.

An excellent example of super-manoeuverability is demonstrated by the Russian Sukhoi SU-27 Flanker jet fighter. It surprisingly demonstrated its agility at the Farnborough air show in 1990,

known as the Cobra manoeuvre. With air speed reduced to a speed of 270 MPH (235 KTS) during straight and level flight, considerable nose-up pitch is attained as the aircraft rears up to extreme high angles of attack. The underside of the aircraft is presented to the oncoming airflow creating considerable lift and drag – the aircraft nearly stops in flight before the nose pitches down to a normal flight attitude.

The experimental Rockwell/DASA X-31 aircraft was built to research super-maneuvrability for fighter type aircraft.

Also known as unsteady aerodynamics.

**Supersonic airflow** A supersonic airflow consists of the airflow being at or greater than the speed of sound over any part of the aircraft.

An aircraft flying faster than Mach 1.0 experience supersonic airflow. Supersonic airflow behaves very different to a subsonic airflow, because pressure waves cannot project ahead of the aircraft to warn the approaching air molecules of the aircraft's impending arrival, as they do in a subsonic flow. Shockwaves, expansion waves and compression waves are always resident in a compressible, supersonic flow; in fact, they begin to make their presence felt at any speed above the aircraft's critical Mach number, where the local airflow-speed over the upper surface of the wing has reached Mach 1.0.

The airflow properties change through the shockwave resulting in an abrupt increase in temperature, density, and pressure with a decrease in the airflow velocity behind the shockwave.

The initial research work of the lift and drag on supersonic airfoils was conducted by Jacob Ackeret (1898–1981).

See **Expansion flow & fan; Blunt nose re-entry vehicle; Superaerodynamics; Diagram 74, Shockwaves.**

**Supersonic airfoils** Supersonic airfoils require different design criteria to subsonic airfoils to optimize lift for high-speed flight. However, their low-speed characteristics are inferior to subsonic airfoils.

The airfoil shape is designed with a bi-convex (symmetrical) low thickness/chord, laminar flow section with the point of max thickness located well aft to produce a favourable pressure gradient over as much of the wing as possible, to minimize drag. Aircraft flying in the low supersonic region below Mach 1.5 may have a leading edge that is fairly well rounded. Double wedge airfoils are favored by aircraft, which operate at very higher Mach numbers, such as research aircraft; the North American Aviation X-15 is one such aircraft.

A leading edge shockwave is present due to the sharp angle of the wing's leading edge, with supersonic flow over all the wing surface, ending in an expansion fan, or fishtail shockwave at the trailing edge. A leading edge shockwave is preferable to a bow shock at low supersonic speeds due to increased wave drag caused by a blunt leading edge; bow shocks are more favourable at higher Mach numbers to dissipate friction heat. At supersonic speeds above Mach 1.3 the aerodynamic drag reduces considerably compared to the drag encountered during acceleration through the transonic region. A supersonic wing's drag coefficient is determined by the skin friction, drag due to lift, and wave drag.

It is commonly assumed all supersonic aircraft would have either sweptback or forward sweep or delta wings, but this is not necessarily so, a short span, straight wing is quite suitable for speeds up to about Mach 1.8. A short span wing does not reach into the shockwave cone emanating from the aircraft's nose. Some early supersonic jet fighters had short span wings – the Lockheed F-104 Starfighter has a wingspan of only 6.62 meters (21 feet 9 inches) for example and the Bell X-1 research aircraft has a span 8.5 meters (28 feet).

See **Supersonic airflow**.

**Supersonic propellers** High propeller tip speeds result in separation of the airflow boundary layer over the blades, causing excessive noise and a loss in efficiency due to compressibility effects. The problems of high tip speed were investigated as far back as 1949–58, in the USA and Europe where research was carried out on propellers designed to operate within supersonic airflow. Curtiss Electric built the first supersonic propeller to fly. The four-blade 3.05 meter (10 feet) propeller turned at supersonic speed and was powered by an Allison XT-38 turboprop engine mounted in the nose of a McDonnell XF-88B Voodoo prototype escort fighter. The first test flight took place on the 14 April 1953.

Research into supersonic propellers was performed at the US Air Forces Propeller Laboratory at the Wright-Patterson AFB in Dayton, Ohio. The flying program continued at Edwards Air Force



**Photo:** The Republic XF-84H Thunderstreak used to flight test a supersonic propeller, lays at peace in the National Museum of the United States Air Force at Dayton, Ohio.

Base in 1955, under the direction of John P. Stack. A modified Republic Thunderstreak, the XF-84H, was used to flight test the four-blade Aero Products supersonic propeller (Mach 1.18 tip speed) with a first flight on 22 July 1955. An Allison XT-40-A-1 5850 ESHP turboprop powered the supersonic prop. The propeller was adorned with a very large spinner to mate with the aircraft's nose, giving the propeller blades a short stubby appearance despite their 3.64 meter (11 feet) diameter. The flight test program was terminated due to problems with the propeller and engine and combined with the inaudible, hypersonic shock waves emanating from the propeller during ground running tests, caused nausea to nearby ground personnel. In fact, when the plane was on the ground the propeller was so noisy it could be heard from 25 miles away! Hence its nickname of 'Thunderscreech' or 'Mighty Ear Banger'.

Several million dollars were spent on the project before supersonic propellers were deemed impractical due to their high noise level and loss of efficiency at high operating speeds. The research was eventually cancelled.

**Supersonic stability** An aircraft increasing speed into the supersonic region will experience a change to its stability and aerodynamic damping.

Accelerating through the transonic speed range, the static longitudinal stability increases, as opposed to the dynamic stability, which reduces. The increase in static longitudinal stability is due the rearward shift of the aerodynamic centre from about 25% MAC at subsonic speeds to about 50% MAC at supersonic speeds. The adverse effect is a decrease in aircraft controllability leading to possible Mach tuck problems. On the other hand, the static and dynamic directional stability both decrease with speeds greater than Mach 1.0 due to the vertical tail lift decreasing and having less effect.

Yaw dampers and other stabilizing systems are required on high-speed aircraft due to reduced aerodynamic damping moments in all three axes, because of the changes in angle of attack of the tailplane with pitch, roll, and yaw and due to reduced density at high altitudes. At 40000 feet altitude the air density is a quarter that of sea level, therefore aerodynamic damping is reduced, decreasing in proportion to the square of the density ratio ( $\sigma$  sigma).

Aeroelastic effects due to high Mach numbers may deform wing or tailplane structures, which can reduce aerodynamic damping and static stability.

**Supersonic tunnel** A wind tunnel designed to operate with a supersonic airflow for high-speed research.

**Super-stall** A super-stall is more of a problem on rear-engine aircraft with T-tails. When the aircraft approaches a stalled condition, the nose will pitch-up further into the stall as opposed to a normal pitch-down. The pitch-up will place the tailplane in the turbulent wake of the main wings, degrading its elevator effectiveness, thus reducing its ability to recover from the stall. With the aircraft in a deep, or super-stall, the loss in lift will start the aircraft on a descending flight path, which in turn will increase the stall angle further. This is the reason for stick shakers and stick pushers being standard equipment on modern jet transport aircraft.

Also known as deep stall or locked-in stall.

**Surface attachment** *See Coanda effect.*

**Surface friction drag** Air resistance due to shearing of the boundary layer close to the surface causes surface friction drag. A very smooth surface can reduce the drag. Surface friction drag plus form drag combine to produce profile drag.

*See Drag; Skin friction drag.*

**Sweepback angle ( $\Lambda$ )** The sweepback angle is the angle between the 25% MAC line (or sometimes the leading edge of the wing) and the aircraft's OY lateral axis. The Greek capital Lambda  $\Lambda$  is used to denote the sweepback angle.

One advantage of an increase in sweepback angle is a resulting decrease in the wing's effective thickness/chord ratio. The physical wing depth remains unchanged, but the effective depth of the wing is less, which has the advantage of a higher critical Mach number. Conversely, too much sweepback angle can result in a partial spanwise flow towards the wingtips causing aileron reversal, aerodynamic tip stalling, shock stalling and wing deformation.

See **Sweptback wings; Diagram 81, Sweepback angle & Critical Mach Number.**

**Sweep-forward angle** The sweep-forward angle is the angle between the OY lateral axis and the leading edge of a swept forward wing with constant chord. If the wing has taper, it is measured similarly except the angle is taken between the OY lateral axis and the wing's 25% MAC line.

The term swept-forward angle is incorrect – use sweep-forward angle or forward-sweep angle.

See **Forward sweep wings.**

**Sweptback wings** The advantages of sweptback wings are found firstly in a higher critical Mach number and drag divergence Mach number and secondly, an increase in lateral stability.

The critical Mach number is higher on sweptback wings compared to straight wings, due to the angle at which the wing meets the on-coming airflow. Increasing the critical Mach number allows the aircraft to fly at a higher cruise speed due to the delayed onset of the wave drag rise. This is best described with reference to **Diagram 81, Sweepback angle & Critical Mach Number.** Vector  $V_0$  represents the free stream airflow over the wing due to forward flight at the aircraft's cruise Mach number. Due to the

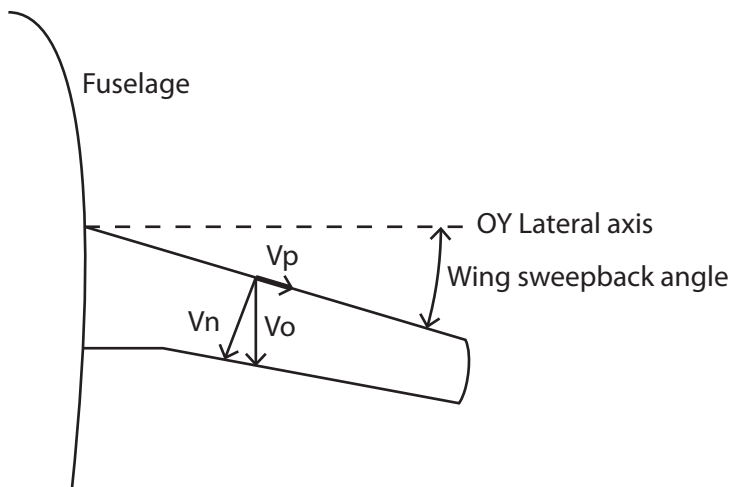
wing's sweepback angle, the vector  $V_n$  normal to the leading edge represents the effective airflow over the wing, which being less than vector  $V_o$ , means the velocity normal to the wing is also less. This is shown by the following for example:

Given: Sweepback angle =  $35^\circ$  ( $\cosine\ 35^\circ = 0.8191$ )

Cruise speed = Mach 0.88

Effective airflow velocity = Mach 0.88 times 0.8191  
( $35 \cos$ ) = Mach 0.80.

Therefore, although the aircraft is cruising at a true speed of Mach 0.88, the wing sees the airflow as Mach 0.80. Due to the effective airflow along the vector  $V_n$ , the wing's geometric thickness/chord ratio appears to the airflow to be less than it actually is. This results in less disturbance to the airflow and an increased critical Mach number producing a better lift grading curve. Increasing the sweepback angle and the use of reduced thickness/chord ratios, both have an effect on reducing the drag coefficient, raising the critical Mach number and delaying compressibility effects, especially at speeds in excess of Mach 1.0. A much higher value of



**Diagram 81, Sweepback angle & Critical Mach Number**



$V_o$  can be flown before  $V_n$  reaches Mach 1.0, and so delaying the formation of Shockwaves.

Vector  $V_p$  represents the spanwise airflow towards the wingtips – an adverse effect on sweptback wings caused by an adverse pressure gradient with boundary layer separation along the span. The energy in the spanwise flow is decreased towards the wingtips leading to premature tip stalling, less lift, less stability and less aileron effectiveness. The spanwise flow towards the wingtips also encourages the airflow to transition from a laminar flow to a turbulent flow. This is the reason for flow fences on sweptback wings – to reduce spanwise flow.

The flaps on sweptback wings produce a lower lift coefficient increase than the same flaps on a straight wing, due to the airflow crossing the flaps at an angle as it does on the main part of a sweptback wing. The reason for unsweeping the wing root trailing edge to be parallel to the OY lateral axis is to improve the direction of the airflow over the inboard flaps to make them more effective.



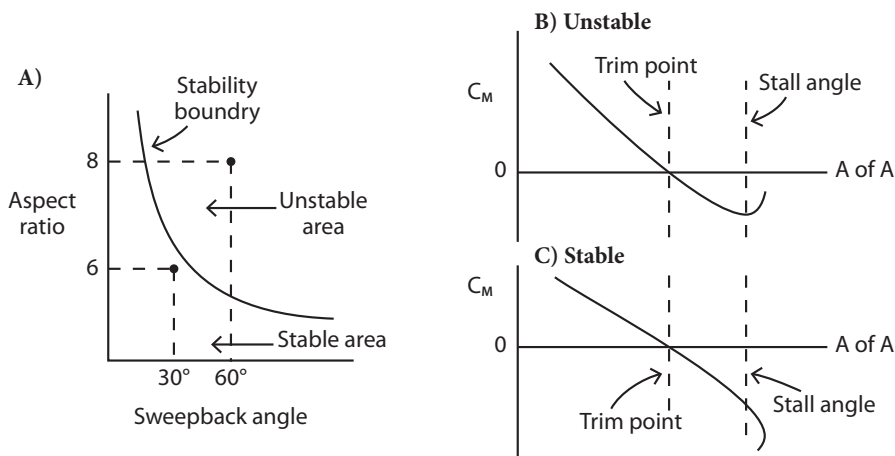
**Photo:** The success of modern jet airliners is due partly to their sweptback wings. The plane in this photograph is a Boeing 757.

A sweptback wing has a lower lift coefficient for a given angle of attack compared to a straight wing aircraft. Therefore, the approach to land (or any slow flight) is in a nose high attitude as opposed to a straight wing aircraft, which has a lower nose attitude.

Wing sweepback angle and the aerodynamic thickness/chord ratio are closely related and together they determine the level of drag rise in the transonic region. The sweepback angle and aspect ratio also have a great influence on the performance and pitch stability of sweptback wings. The degree of sweepback angle must remain within certain limits for a given aspect ratio. For example, an aircraft with an aspect ratio of eight and a sweepback angle of  $60^\circ$  is shown as point 1 on **Diagram 82, Sweepback Angle & Aspect Ratio**. This area of the graph represents the unstable region, where an aircraft would experience a nose pitch-up at the stall. Reducing the wing aspect ratio to six with a sweepback angle of  $30^\circ$  for example, results in a stable aircraft. This is shown by the area as point 2 on **Diagram 82A**. The stability boundary on the graph dividing the stable and unstable regions is approximate and varies depending on the degree of taper ratio.

On **Diagram 82B and C** the sloping line represents the pitching moment ( $C_M$ ) indicating positive stability. **Diagram 82B** shows how the stability changes from positive to negative when the aircraft enters a stall. The positive stability line turning upwards at its lower end shows this. **Diagram 82C** shows the positive slope turning downwards indicating an increase in stability.

Another advantage of sweptback wings is their inherent lateral stability. A sweptback wing induces a lateral restoring moment to right the aircraft when the aircraft rolls in turbulence. Consider a roll to the left; the aircraft will sideslip towards the lower (left) wing. The left wing will present a greater thickness/chord ratio with a greater effective span on the left wing. Because the fuselage shields part of the right wing near the root, its effective lifting area is reduced compared to the left wing. Therefore, the left wing produces more lift and less drag with the opposite effect on the right wing. The imbalance of lift and drag on each wing tends to restore the aircraft back to straight and level flight. A sweptback wing has the same effect on lateral stability as does dihedral effect.



**Diagram 82, Sweepback Angle & Aspect Ratio**

John William Dunne (1875–1949) was an Irish aeronautical engineer and he is credited for inventing and designing the first swept wing aircraft, the tailless Dunne D.1 glider and other sweptback wing aircraft, which followed. Dunne used sweptback wings for improved lateral stability. The sweptback wing fell out of favor until being 're-invented' by Doctor Adolph Büsemann (1901–86) in WWII also for the purpose of lateral stability in high-speed flight. However, it was not until 1945 when Robert Thomas Jones (1910–1999) a NACA aerodynamicist, suggested using the sweptback wing to raise the critical Mach number. Since then, swept-back or delta wings, have always been used on high-speed jet aircraft.

NASA flew a converted piston-engine Bell P-63 Kingcobra WWII fighter, re-designated as the Bell L-39, with sweptback wings installed. The flight-testing in 1947 was required to investigate the stability, control, and handling characteristics during low-speed flight of wings swept back to 35 degrees. The North American F-86 Sabre jet fighter was the first successful production, sweptback wing aircraft to fly.

**See Rams horn vortex; Lift coefficient & sweptback wings.**

**Swept wing chord** The chord of a swept wing aircraft is measured across the width of the wing, parallel to the aircraft's centreline and is the average of the tip and root chord distance.

**Swing-wing aircraft** The swing-wing concept was initially developed by Messerschmitt in Germany during WWII and further developed by the English engineer Barnes Wallis (1887 – 1979) as early as 1945.

Swing-wing is an alternate name for an aircraft with a variable sweepback wing. Having variable sweepback angles allows the aircraft to fly with the wings in the most aerodynamically efficient position, which is straight wings for take-off and landing, medium sweep wings for high subsonic cruise and fully swept for supersonic flight.



**Photo:** The Rockwell B-1B Lancer is a swing-wing, supersonic bomber built for the US Air Force. It is on display at the National Museum of the US Air Force in Dayton, Ohio.

**Symbols** Several Greek letters, or symbols, are used in aerodynamic theory to represent various terms. Some of the more common symbols in use are listed below:

|                                               |                                                                                                       |
|-----------------------------------------------|-------------------------------------------------------------------------------------------------------|
| <b>Alpha <math>\alpha</math>:</b>             | angle of attack (geometric); angular and linear acceleration;                                         |
| <b>Beta <math>\beta</math>:</b>               | angle of sideslip; propeller pitch angle; tailplane angle of incidence; propeller blade A of A        |
| <b>Delta <math>\delta</math>:</b>             | angle of control surface deflection; static pressure; boundary layer thickness; vortex sheet strength |
| <b>Epsilon <math>\epsilon</math>:</b>         | angle of downwash                                                                                     |
| <b>Eta <math>\eta</math>:</b>                 | efficiency; propulsive efficiency; elevator angle                                                     |
| <b>Gamma (capital) <math>\Gamma</math>:</b>   | vortex strength; circulation                                                                          |
| <b>Gamma <math>\gamma</math>:</b>             | dihedral angle; gliding angle; angle of climb                                                         |
| <b>Kappa <math>\kappa</math>:</b>             | compressibility                                                                                       |
| <b>Lambda (capital) <math>\Lambda</math>:</b> | angle of sweepback                                                                                    |
| <b>Lambda <math>\lambda</math>:</b>           | wing area ratio $S/b^2$ ; damping coefficient; scale ratio                                            |
| <b>Mu <math>\mu</math>:</b>                   | viscosity of fluid; coefficient of friction; Mu ratio                                                 |
| <b>Nu <math>\nu</math>:</b>                   | kinematic viscosity                                                                                   |
| <b>Omega <math>\omega</math>:</b>             | angular velocity                                                                                      |
| <b>Phi <math>\phi</math>:</b>                 | angle of bank or roll; effective helix angle; pitch; entropy                                          |
| <b>Pi <math>\pi</math>:</b>                   | mathematical constant of 3.142; circumference to diameter ratio.                                      |
| <b>Psi <math>\psi</math>:</b>                 | angle of yaw; fin sweep angle (vertical fin); stream function                                         |
| <b>Q:</b>                                     | mass of flow (through jet engine)                                                                     |
| <b>Rho <math>\rho</math>:</b>                 | air density                                                                                           |
| <b>Sigma <math>\sigma</math>:</b>             | relative air density; density ratio; Prandtl's interference factor (biplanes)                         |

|                                             |                                                                                |
|---------------------------------------------|--------------------------------------------------------------------------------|
| <b>Theta (capital) <math>\Theta</math>:</b> | absolute temperature                                                           |
| <b>Theta <math>\theta</math>:</b>           | angle of pitch; temperature; Mach cone<br>semi-vertex angle; temperature ratio |
| <b>Xi <math>\xi</math>:</b>                 | aileron angle                                                                  |
| <b>Zeta <math>\zeta</math>:</b>             | rudder angle.                                                                  |

**Symmetrical airfoil** A symmetrical airfoil has no camber, the curvature is the same on both sides of the mean line; the chord and camber line coincide. The vertical tail is a symmetrical airfoil with both sides of equal camber. However, on most aircraft (aerobats can be an exception) the main wing is cambered, that is the upper surface has greater curvature than the bottom surface. Propeller blades and some helicopter rotor blades are similarly shaped.

Examination of the familiar lift coefficient versus angle of attack graph **Diagram 49, Lift Coefficient V. Angle of Attack**, we find that the symmetrical airfoil ceases to produce lift at zero degrees angle of attack. [The use of symmetrical airfoils can be an advantage on helicopter main rotors due to the following:

- light weight materials can be used in their construction
- they produce less drag than cambered airfoils
- the centre of pressure and aerodynamic centre are both, theoretically, located on or near the 25% MAC (quarter chord) location
- Locating the aerodynamic centre, centre of pressure and centre of gravity on or near the blade's feathering axis reduces blade stress due to less twisting moments with changes in angles of attack.

However, one disadvantage is:

- symmetrical airfoils have relatively sharp leading edges, which tends to lower the critical angle and therefore reduce the maximum lift coefficient.

See **Directional static stability; Rotor blades; Feathering axis.**

**Symmetric stall** A stall condition where both main wings stall evenly in a level flight attitude with the nose dropping without any yaw involved is known as a symmetrical stall.

Also known as a square stall.

**Symmetry of lift** A condition where both (or all) helicopter rotor blades are producing equal proportions of lift force.

*Compare with* **Dissymmetry of lift.**

**Synchropter** *See* **Main rotor.**

# T

**Tabs** Tabs are small, auxiliary control surfaces hinged to the trailing edge of primary control surfaces.

The purpose of the trim tab is to aid the movement of the primary control by reducing or increasing the hinge moment and enhancing the control power. This can be achieved by using leading or lagging tabs where a linkage system is attached between the fixed control surface and the moveable trimming tab.

The leading tab is geared to displace upwards and assist the control surface to rise resulting in an increase in control surface hinge moment. The lagging tab works in the opposite sense to the leading tab. The lagging tab, also attached to the fixed tail surface, deflects downwards with up-elevator to help raise it and to reduce the hinge moment.

Tabs may be of the anti-balance tab, balance tab, fixed tab, or servo tab, anti-servo tab or spring tab.

**Tactical turning performance** *See* Minimum radius of turn.

**Tail** The aircraft's tail (or tailplane) is located at the aft end of the fuselage and consists of the vertical fin, the horizontal stabilizer wing, or a butterfly tail. Its purpose is to provide a stabilizing force and a means to control the aircraft around the pitch and yaw axes.

*See* Tailplane.

**Tail arm** The distance between the aircraft's centre of gravity and the centre of pressure of the horizontal tailplane determines the tail arm. The greater the distance, the greater the force produced for a given control deflection and air speed.

It is also known as the tailplane moment arm.

*See* Tail volume coefficient (V).



**Tail boom** Tail booms are mounted on helicopters to support the tail rotor at a convenient distance from the centre of gravity to provide yaw control and to oppose the main rotor torque.

On airplanes, tail booms are used mostly in pairs, protruding from the main wing's trailing edge to support the tailplane. They may, or may not be a continuation of the engine nacelles. Some examples include the Lockheed P-38 Lightning fighter of WWII fame, the Hawker Siddeley Series 222 Argosy freighter, the deHavilland DH.100 Vampire jet fighter, and the Cessna 336 Skymaster. In addition, in more recent times the Adam A500 push/pull piston-twin has tail booms, as also does the Burt Rutan designed suborbital SpaceShipOne space plane and its mother ship White Knight, to mention just a few examples.



**Photo:** The Fairchild C-119J freighter must be one of the largest aircraft in the world to be equipped with a twin tail boom. This one is parked in the Air Park at the National Museum of the US Air Force in Dayton, Ohio.

**Tail cone** A conical shaped fairing cone is located at the far rear of the fuselage where the fuselage radius reduces to a point. Jet transports commonly have their auxiliary power unit (APU) mounted in the tail cone area.

A second type of tail cone is located within the exhaust section of a gas turbine (jet) engine to help smooth the flow of the exhaust.

**Tail efficiency factor** The tail efficiency factor is an indicator of the tail's efficiency determined by its location relative to the airplane's wake from the main wings, the slipstream from the airflow passing over the aircraft and in the case of piston-engine aircraft, the propwash. Careful design can provide a tailplane that is 100% efficient and tailored to the aircraft's static stability.

See **Tail load; Tailplane.**

**Taileron** A taileron is a slab tailplane mounted so that the left and right surfaces may act in unison to provide the required pitching moment or in opposition as ailerons for roll control. Some high-speed jet fighter aircraft have a taileron in place of ailerons, because ailerons could place too much bending stress on the main wings. The McDonnell Douglas Phantom II jet fighter is one example that comes to mind.

*Compare with **Elevons**, which are hinged to the trailing edge of (usually) delta wings that perform the same operation of pitch and roll.*



**Photo:** The General Dynamics F-111 fighter/bomber has a slab tailplane, or taileron to be correct. This F-111 is on splay at RAF Museum Cosford, UK.

**Tailets** Tailets are additional, small vertical tails attached to the horizontal tail to provide additional yaw stability. They are found for example on the T-tail of the Beech 1900D commuter aircraft below the horizontal tailplane.

**Tail fin** The tail fin is the vertical airfoil and part of the empennage for providing stability and control in the yawing plane.

*See Tailplane.*

**Tail-first airplane** This is a common, but incorrect alternate name for a canard equipped airplane.

*See Canard wings.*

**Tail-heavy** A tail-heavy aircraft is one that is tending towards an out-of-balance condition where the centre of gravity is too far aft, towards the rear of the aircraft. Tail-heavy aircraft do not fly in a tail-down and nose-high attitude – a level attitude is maintained by forward use of the elevators. However, an aircraft may sit tail-low on the ground.

See **Tail load**.

**Tailless aircraft** The tailless aircraft is one devoid of any empennage, or tail unit.

A few aircraft with delta wing configuration are also tailless types. Some sweptback wing, tailless aircraft have proven to be less successful; although there are a few advantages with the tailless configuration, there are also several disadvantage. The advantages are listed as follows:

- greater manoeuvrability
- reduced drag and weight
- No tail unit to build means no cost.

The disadvantages of the tailless aircraft are:

- the location of the control surfaces are required to be as far as possible from the centre of gravity
- reduced stability and control effectiveness due to the short moment arm
- greater control deflection is required to instigate a change in attitude; causing extra trim drag
- The short moment arm requires greater control surface area.

The Englishman John William Dunne (1875 – 1949) introduced the first tailless gliders as early as 1907. The Armstrong Whitworth A.W.52 was the first tailless powered aircraft.

**Tail load** In order to balance the forces acting on the main wing during flight, the tailplane either produces an up or download. Generally, during straight and level flight the load on the tailplane is a download, which counteracts the aircraft's nose down pitching moment. If a tail download exists (which may be up to 5% of the aircraft's weight) the main wings have to provide additional lift to counteract it, which will raise the stall speed slightly. A zero tail load is ideal, when it can be achieved.

Some helicopters have an inverted, cambered airfoil tailplane to provide a constant tail-down force during flight to balance the nose-down pitching force.

An all-flying tailplane can be trimmed to align the angle of incidence with the streamlined airflow, to reduce trim drag.

See **Forces in cruise flight – airplane; Tail-heavy.**

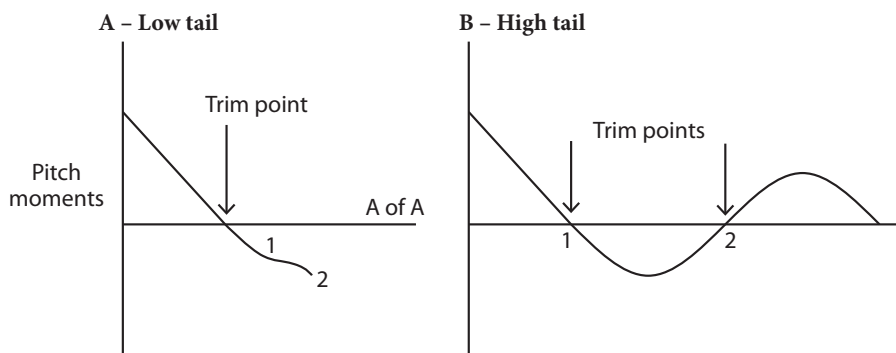
**Tailplane** A tailplane may be a fixed surface with elevators, or a one-piece variable incidence type. The reasons for a tailplane are:

- to provide control in the pitching plane about the OY lateral axis
- to provide stability and oscillation damping in the pitching plane to counteract the instability of the main wing
- to balance the forces involved due to changes in speed, trim and centre of gravity
- To counteract the change in forces on the main wing due to flap and leading edge device application.

Tailplanes are either cambered or a symmetrical airfoil shape with zero degree angle of attack when in the trim condition. It may be a non-lifting surface under normal load conditions, although it is quite common for the tailplane to provide a download during trimmed flight. However, as long as the tailplane can provide a pitching moment greater than the main wing upsetting moment, the aircraft will be longitudinally stable.

With a reducing flight speed, the angle of attack increases and the centre of pressure moves forward on the main wing. The centre

of pressure on the tailplane also moves forward and its lift increases by a small amount. However, due to its distance from the centre of gravity and its longitudinal dihedral, the tailplane produces a nose-down restoring moment to lower the nose back to the trim flight speed. Larger surface tailplanes also provide a greater lift/stabilizing moment. It can be seen, any change in the aircraft's trim speed or centre of gravity, will require a restoring moment from the tailplane.



**Diagram 83, Tailplane & Trim Point**

**Diagram 83, Tailplane & Trim Point**, shows the graph plotted for an aircraft with a low tail position. The down slope represents a stable aircraft. As the angle of attack increases (moving to the right across the horizontal axis) the restoring moment is positive and acts to lower the nose. Point 1 on the diagram represents a high angle of attack of a low tailplane immersed in the main wing's wake where the elevator effectiveness is reduced. As the angle of attack increases further (to point 2 on the diagram), the main wing's wake passes above the tailplane and the tailplane's stability improves (shown by the slope curving downward again after leveling off). Where the aircraft's positive down slope crosses the horizontal is the trim point.

**Diagram 83B** is drawn for an aircraft with a high tail position, such as a T-tail design. The diagram is initially the same as that shown for 83A except for the shape of the curve below the horizontal and not one, but two trim points at positions A and 2.



**Photo:** An Air New Zealand/Link Beech 1900D on short finals to Wellington Airport. Note the appendages on the tailplane – a T-tail with taillets. Delta fins are located on the lower fuselage below the T-tail.

With increased angle of attack, the main wing wake becomes broader with less energy and dynamic pressure. Stability is reduced considerably as the angle of attack increases further when the tail passes through the main wing's wake – this is shown by the curve rising up above the horizontal line (position 1). When the main wing angle of attack is at a position where its wake passes above the T-tail, positive longitudinal stability returns and the second trim point is reached (position 2). However, the aircraft by then is in a deep stall at a very high angle of attack and if the elevator has run out of nose-down travel, recovery from the deep stall will be

impossible. This is the reason for stick shakers and stick pushers on jet transport aircraft – to prevent the aircraft from entering a deep stall condition in the first place. Aft fuselage-mounted engines can aggravate the wake over the tail.

T-tails are commonly used on some jet transport aircraft, business jets and a few piston-engines types. However, jet fighter aircraft favor low tail positions where an incipient stall will occur before any chance of a total loss of control due to a deep stall occurring. Sweptback vertical tails on light aircraft are less efficient than straight tails but many pilots for aesthetic reasons favor them. The tailplane are of several different designs. These include the following:

- conventional tail with the horizontal tail attached to the side of the fuselage and with the vertical fin on top
- the V-tail
- the cruciform tail
- the twin or H-tail
- the T-tail
- The triple tail.

It is known as the tail-unit in the US terminology and also horizontal stabilizer or all-flying tail.

*See Moment arm; Empennage; Longitudinal dihedral; Tail; Tail load; Diagram 61, Pitching Moments V. Angle of Attack.*

**Tail rotor** The tail rotor is an anti-torque propeller found on most helicopters; the McDonnell Douglas 520 NOTAR helicopter is an exception. The tail rotor counteracts the main rotor torque and provides control in the yawing plane.

A helicopter with anti-clockwise rotating main rotor blades usually has its tail rotor mounted on the left side, where it pushes the tail boom to the right to counteract main rotor torque by ejecting outflow air from the left-hand side of the tail rotor. Therefore, tail rotor thrust is towards the right to produce an equal



and opposite force to counteract main rotor torque. The greater the distance of the tail rotor axis from the main rotor axis determines the amount of anti-torque force provided. Varying the tail rotor thrust is achieved by changing the rotor blade pitch angle (by use of the rudder pedals, or anti-torque pedals, as they are correctly known) to rotate the helicopter around its OZ normal axis.



**Photo:** A Russian Mil-8 helicopter with a twin-turbine powered, five-blade, clockwise rotating main rotor, and pylon-mounted pusher tail rotor on the right side of the boom.

To turn/yaw the helicopter to the right requires less engine power from the tail rotor to reduce its pushing thrust; main rotor torque then overcomes tail rotor thrust and the helicopter yaws right. Therefore, with less tail rotor power required, the surplus power is transferred to the main rotor allowing engine RPM to increase along with an increase in lift and the helicopter rises. Conversely, a turn to the left requires more power to increase the thrust of the tail rotor in order to push the tail boom to the right, which

deprives the main rotor of some of its power. The main rotor RPM then decreases along with the lift and the helicopter sinks. Pilot control input is required to maintain a constant height whenever the helicopter is yawed left or right in the hover.

On modern helicopters, whichever way the main rotors rotate, the tail rotor is on the same side of the fuselage as the main rotor's retreating blade, where it pushes the tail boom to counteract main rotor torque. If the tail rotor is mounted on the side of the advancing main rotor blade, the tail rotor then pulls the tail boom to counteract main rotor torque; it is then termed a 'puller'. An inspection of photographs of very early helicopters reveals their tail rotors to be mounted on the opposite side of the tail boom to modern helicopters; early helicopter tail rotors were 'pullers' while their modern counterparts are mostly 'pushers'.

The tail rotor's airfoils sections are generally of non-symmetrical shape as opposed to the main rotor's airfoil, which may or may not be symmetrical. The smaller diameter of the tail rotor enables it to withstand the associated stress forces inherent in cambered wings.

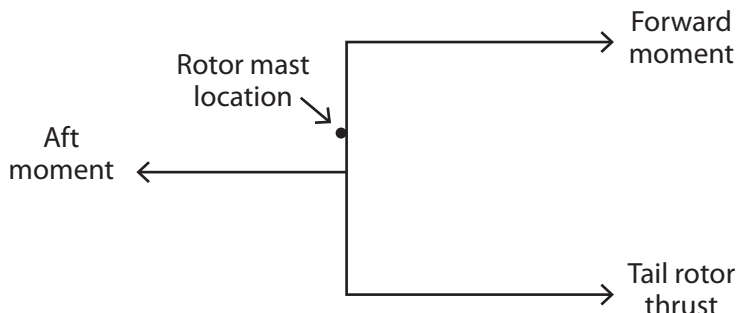
The tail rotor was first used by Igor Sikorsky (1889–1972) on his VS-300 helicopter (first flight 1939), which has become the most common form of anti-torque control system for most single rotor helicopters.

*See Reaction torque.*

**Tail rotor drift** Tail rotor drift becomes an operational problem when the helicopter is in hovering or low-speed forward flight.

With the main rotor turning anti-clockwise, a torque couple is established that attempts to turn the fuselage clockwise around the rotor mast. The torque component acting to the rear of the main rotor mast attempts to pull the helicopter fuselage towards the left while the front component of the torque is acting towards the right-hand side, the same side as the tail rotor thrust is acting. The combined forces from the main rotor and the tail rotor attempts to pull the helicopter to the right – the helicopter experiences tail

rotor drift (to the right). **Diagram 84, Tail Rotor Drift**, shows this. Tail rotor drift would be to the left with the main rotor turning clockwise.



**Diagram 84, Tail Rotor Drift**

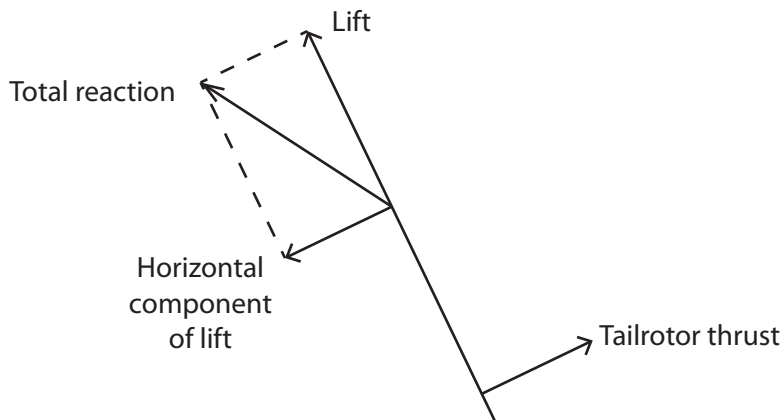
With increasing forward speed, the helicopter's vertical fin becomes more effective in providing directional stability and less work is required from the tail rotor with the advantage tail rotor drift is substantially reduced. Tail rotor drift is countered by means of tilting the main rotor disc to provide a small sideways force in opposition.

It is also known as translating tendency or anti-torque drift.

**Tail rotor roll** Correcting for tail rotor drift requires the helicopter's main rotor disc to be tilted to the left, assuming an anti-clockwise rotating disc.

The high horizontal component of the main rotor, total rotor thrust acts to the left. Coupled with the opposing low horizontal thrust from the tail rotor acting to the right, a rolling couple is produced, which tends to roll the helicopter to the left. **Diagram 85, Tail Rotor Roll**, shows this.

One method to reduce tail rotor roll is to raise the tail rotor on a pylon at the end of the tail boom, thus reducing the moment arm between the main and tail rotors, as found for example on the Bolkow BK 117 helicopter.



**Diagram 85, Tail Rotor Roll**

**Tail sitter** The Convair XF-1 Pogo and Ryan X-13 Vertijet were deemed tail sitters because they were designed to take-off vertically (like a rocket) from a tail sitting position. The concept proved to be too impracticable especially for the pilot while flying backwards to land in the tail sitting position!



**Photo:** The Ryan X-13 Vertijet on display at the National Museum of the United States Air Force at Dayton, Ohio.

**Tail volume coefficient (V)** The tail volume coefficient is a design requirement to determine the required tailplane size and its required location distance (lever arm) from the main wing's centre of gravity and includes the wing's span, area and the mean aerodynamic chord (MAC). Aircraft with higher volume coefficients have greater stability in the pitching plane with a greater CG range; however, large tailplane area also produces greater drag.

The horizontal tail volume coefficient is determined by the following formula:

$$V_H = S_H \times L_H / S_W \times MAC$$

Where  $V_H$  = tail volume coefficient

$S_H$  = horizontal tail area

$L_H$  = distance between tail's  
a.c. and aircraft CG

$S_W$  = main wing area

MAC = mean aerodynamic chord.

The vertical tail volume coefficient has a similar formula thus:

$$V_V = S_V \times L_V / S_W \times b$$

Where  $S_V$  = vertical tail area

$L_V$  = distance from the vertical tail's  
aerodynamic centre to the aircraft c.g.

$S_W$  = main wing area

$b$  = wingspan.

**Tandem wing** A tandem wing aircraft has two sets of main lifting wings located forward and aft on the fuselage. Fore and aft stabilizers do not count as tandem wings and neither do biplanes.

The advantage of tandem wings is more wing area is available for creating lift. The disadvantage is the greater weight of the structure and the additional drag, which requires more engine power. Although, some modern tandem wing aircraft are made of composite materials, which has the advantage of less weight and a

smoother surface to cut down friction drag and hence, less power is required.

Because the centre of gravity is located mid-way between the two tandem wings, the aerodynamics involved are different to that found on a single-lifting airfoil. As the aircraft accelerates in flight, the angle of attack reduces and the centre of pressure moves aft on each wing. The centre of pressure on the forward wing moves closer to the centre of gravity as opposed to the rear wing where the centre of pressure moves away from it. The result is the aircraft pitches down with an increase in speed, which requires a greater load from the elevators to counteract it. Trim drag is also increased.

One of the first tandem wing aircraft was the Frenchman Henri Mignet's (1893–1965) HM.14 Flying Flea, which first flew in 1933. Another example was the Delanne Duo-Mono also designed by a



**Photo:** Henri Mignet's Flying Flea, was one of the first tandem wing aircraft to fly. This one is on display at the Royal Air Force Museum at Cosford, England.

Frenchman, Maurice Delanne; its first flight occurred circa 1937–8. Tandem wing aircraft were not very common until the American Burt Rutan designed the Rutan Quickie in 1977. The Dragonfly, another tandem wing aircraft, and the Quickie have both proven to be very popular with the homebuilder fraternity.

**Tangential velocity** The tangential velocity is the speed of the propeller rotation (RPM) along the plane of rotation shown by vector A-B on *Diagram 68, Propeller Pitch*.

It is an alternate name for rotational velocity.

**Taper** A wing is said to have taper in depth when the wing depth decreases from the roots to the tips, which combined with taper in planform has the advantage of reducing the weight of the wing structure across the span.

Taper in planform is found when the chord decreases in length towards the tips. Planform taper increases the wing's aspect ratio, which in turn decreases the induced drag and provides a lift distribution profile closer to the ideal elliptical curve. A tapered wing has less area ( $S$  in the lift formula) than a comparable wing with constant chord and same span. The taper in planform may have both leading and trailing edges tapering in towards the wingtips, or one edge may be tapered and the other straight. Taper in wing plan and/or thickness reduces the lift coefficient over the outer portions of the wing, with the advantage of reducing wing-bending loads and inducing the wing root area to stall before the tips do. If the taper ratio is too small as on a constant chord wing,



the wing may be too heavy at the wingtips, which could have an adverse effect on rolling and lateral stability. If the taper ratio is too large having very narrow chord tips, then the tips will tend to stall early before the wing roots stall – an undesirable situation.

The wing may also have taper for aesthetic reasons – a straight wing is not very appealing.

*See **Aspect ratio**.*

**Taper ratio** The amount of wing taper ratio ( $\lambda$ ) is measured by the ratio of tip chord ( $C_t$ ) to root chord ( $C_r$ ). It is commonly around 0.4 on a wing with an elliptical lift distribution loading, or for a Piper Cherokee type of rectangular wing the taper ratio is 1.0 and a delta wing has a taper ratio of zero. The taper ratio can be shown by the formula:

$$\text{Taper ratio } (\lambda) = C_t / C_r$$

Where  $C_t$  = tip chord  
 $C_r$  = root chord.

Herman Glauert (1892–1934) the English aerodynamicist was the first person to calculate the effects of a wing's taper ratio.

**Teetering rotor system** A helicopter with a teetering rotor system has two main rotor blades, which are free to pivot about the horizontal transverse axis. The teetering system allows the blades to flap and feather due to the seesaw (teetering) action; this reduces the lead/lag stress forces caused by the Coriolis and centrifugal force effect. Arthur M. Young (1905–1995), while working for Bell Aircraft Company, invented and patented the stabilizer bar, to enhance the stability of the teetering rotor system. It is also known as the semi-rigid rotor system.



**Photo:** This photo shows the two-blade teetering rotor head atop the mast of a piston-engine powered Robinson R44 helicopter.

**Temperature** Temperature is a measure in degrees of how hot or cold an object is; or we can rephrase this statement to read 'the temperature scale is related to the thermal energy in a system'. Based on the molecular theory, the higher the temperature the greater the molecules vibrate, each molecule having greater kinetic energy as the temperature increases. With a drop in temperature, the molecules vibrate less and less until they are assumed virtually motionless at the absolute zero temperature (minus  $273.15^{\circ}\text{C}$ ). Jacques Charles (1746–1823) a French physicist discovered circa 1787 the absolute zero temperature.

In the aviation industry, temperature is relevant in meteorology, engine temperature limitations and aircraft performance, etc. The temperature, given in degrees Celsius ( $^{\circ}\text{C}$ ) is almost universally accepted in the industry and becoming more popular with the public in general, and most countries have now converted from degrees Fahrenheit ( $^{\circ}\text{F}$ ) to Celsius. Moreover, in physics and science, temperature in degrees Kelvin is also used for many scientific calculations. In fluid dynamics, the temperature should be defined as static temperature, to avoid any confusion.

Daniel Gabriel Fahrenheit (1686–1736) a German scientist introduced the Fahrenheit scale in 1714. In 1709, he devised a thermometer containing alcohol and later mercury. The temperatures for fresh water at sea level are:

- absolute zero =  $-460^{\circ}\text{F}$ .
- freezing point =  $32^{\circ}\text{F}$
- boiling point =  $212^{\circ}\text{F}$ .

The Swedish Professor of Astronomy, Anders Celsius (1701–1744) devised the Celsius temperature scale in 1742. The Celsius scale is based on a scale of 100 divisions. The temperatures are:

- absolute zero =  $-273^{\circ}\text{C}$
- freezing point =  $0^{\circ}\text{C}$
- boiling point =  $100^{\circ}\text{C}$ .

Lord Kelvin, Baron William Thomson (1824–1907) a Scottish physicist and mathematician devised the metric Kelvin scale. The Kelvin scale is based on the absolute zero being the theoretical minimum temperature obtainable, equivalent to  $-273.15^{\circ}\text{C}$ . The temperature scale divisions are the same size as found in the Celsius scale.

- absolute zero = zero Kelvin (0K)
- freezing point = 273K
- boiling point = 373K.

In 1865, the Scottish engineer and scientist, William John Macquorn Rankine (1820–1872) introduced his English Rankine temperature scale. His scale is based on one Rankine (1R) being equal to 1 degree Fahrenheit. The capital letter 'R' follows the temperature without a degree sign, similar to the Kelvin scale. The Rankine scale is seldom used these days. The Rankine temperature scale is used for calculations in Charles' Law. The temperatures scale is:

- absolute zero = zero Rankine (0R)
- freezing point = 492R
- boiling point = 672R.

The following is a simple formula for the conversion of temperature between Celsius to Fahrenheit and vice versa:

- Fahrenheit to Celsius:  $x^{\circ}\text{F} - 32 \text{ times } 5/9 = y^{\circ}\text{C}$ .
- Celsius to Fahrenheit:  $x^{\circ}\text{C} \text{ times } 9/5 + 32 = y^{\circ}\text{F}$ .

See **Temperature ratio**.

**Temperature ratio** The temperature ratio is relevant to compressibility effects and jet engine performance. Temperature is given the Greek symbol of  $\theta$  (theta).

$$\text{Temperature ratio} = \frac{\text{Ambient air temperature}}{\text{Standard sea level air temperature}} = T/T_o$$

$$\theta = \frac{^{\circ}\text{C} + 273}{288}$$

**Temperature rise** The formula to find the rise in temperature due to kinetic heating is:

$$\text{Temperature rise } ^{\circ}\text{C} = \text{TAS (MPH/100)}^2$$

Subtract the temperature rise from the outside air temperature gauge to find the ambient temperature.

See **Aerodynamic heating; Kinetic heating; Impact temperature.**

**Test pilots** In the early days of experimental test flying (post WWII) the test pilots involved in flying the new research aircraft became household names. Names such as ‘Chuck’ Yeager, Scott Crossfield, and Milburn Apt, to name just three. Listed below is a brief summary of test pilots who made aviation history. We start with a look at the pilots who were the first to achieve Mach 1, 2, 3, 4, 5, and 6, followed by other famous test pilots.

- **Mach 1:** USAF Major Charles E. ‘Chuck’ Yeager’s (1923–) claim to fame, occurred on 14 October 1947 when he became the first person to exceed the speed of sound when the Bell X-1 he was piloting reached Mach 1.06.
- **Mach 2:** US Navy test pilot Albert Scott Crossfield (1921–2006) was the first pilot to exceed twice the speed of sound when he

flew the Douglas D-558-2 Skyrocket to a speed of Mach 2.005 on 20 November 1953.

- **Mach 3:** Captain Milburn Apt (1924–1956), a US Air Force test pilot flying the Bell X-2 was the first person to exceed Mach 3 when he reached Mach 3.196 (2094 MPH) on 27 September 1956. However, within minutes of his achievement his rocket plane went out of control due to inertia coupling, Captain Apt was knocked unconscious and died in the ensuing crash.
- **Mach 4, 5 and 6:** Captain Robert M. White (1924–2010), also a US Air Force test pilot, was the first to exceed Mach 4, 5 and 6, all in 1961, flying the North American Aviation X-15 research aircraft. On 7 March 1961, he achieved Mach 4.43, followed by Mach 5.27 on 23 June and finally Mach 6.04 (4093 MPH) on 9 November.
- **Mach 6.7:** US Air Force Major William J. ‘Pete’ Knight (1929–2004), achieved the final speed record of Mach 6.7 (4520 MPH) on 3 October 1967 in the North American Aviation X-15.

Other names include:

- Howard Clifton Lilly (1916–1948) the first NACA test pilot to be killed on duty.
- Michael J. Adams (1930–1967)
- Milton O. Thomson (1926–1993)
- Joseph (Joe) Albert Walker (1921–1966)
- William Harvey Dana (1930–20?).

These are just a few of the men with the ‘right stuff’ who flew many missions on the research aircraft operated out of Edwards Air Force base in California. Test pilots who exceeded an altitude of 50 nautical miles above the Earth’s surface were awarded astronaut wings. [Fifty miles altitude is considered the fringe of space]. Joe Walker (1921–1966) was the first pilot to receive astronaut wings for his high altitude flights in the X-15 rocket plane. On 22 August 1963, he set an altitude record of 354 200 feet (67.08 miles) the highest altitude for a non-spacecraft vehicle. On the other side of the Atlantic, British and French test pilots include the following:

- British test pilot Lionel Peter Twiss (1921–?) was Fairey Aviation's chief test pilot from 1946–1959 following his career as a WW II Naval pilot. He piloted the Fairey Delta 2 on the record breaking flight passing 1000 MPH (1131.76 MPH to be precise) or Mach 1.7 on 10 March 1956.
- The test pilots responsible for flight-testing Concorde were the Englishmen Captain Brian Trubshaw (1924–2001) BAC's chief test pilot with co-pilot John Cochrane (1930–2006) and the Director for flight-testing for Aerospatiale, Captain Andre Edouard Turcat (1921–20?) and his co-pilot.
- Captain Eric 'Winkle' Brown (1919–20?) was a Royal Navy pilot during WWII and later became commander and test pilot for the RAe Aerodynamics Flight at Farnborough, UK. He is credited with many firsts in aviation and holds the record for the most number of aircraft types flown, over 487 in total, not counting variants of the same type.
- John Cunningham (1917–2002) was a test pilot for DeHavilland both before and after WWII. He was a distinguished RAF officer and night fighter ace during the war with 20 kills and was instrumental in introducing airborne radar, known then as Airborne Interception. Returning to DeHavilland's after the war, he became chief test pilot following the loss of Geoffrey DeHavilland Jr. John Cunningham went on to flight test the DH Comet jet airliner and the Trident III, amongst other planes.

This short list is by no means complete, many other test pilots also deserve recognition for their exploits and service to the advancement of flight.

See **Research aircraft**.

**Theodorsen T.** Theodore Theodorsen (1897–1978) was born in Norway and later moved to America where he joined NASA as an aerodynamicist. He worked on developing the theory of airframe flutter/aeroelasticity; he used open and closed wind tunnels to investigate the effects of skin friction subsonic, transonic, and supersonic flight. He performed further development work on aircraft propellers, thin airfoil theory and also wing section theory and their pressure distribution and other work in the field of physics.

**Thermal barrier** *See Aerodynamic heating.*

**Thermal boundary layer** The thermal boundary layer is present on high supersonic-speed aircraft and hypersonic re-entry vehicles. The thermal boundary layer is caused by the high-speed friction, which heats the air adjacent to the aircraft's body, (known as the wall temperature). With increasing distance from the body surface, the temperature decreases to be equal to 99% of the freestream ambient air temperature; this is the thermal boundary layer thickness. It is superimposed on and extends further than the familiar velocity boundary layer.

**Thermal cycle** *See Brayton cycle.*

**Thermal soaking/thicket** The term thermal soaking or thicket was introduced by NACA in the early 1950s when early research aircraft such as the Bell X-1 and X-1A to D models were designed and used to fly at speeds exceeding Mach 2.0 and altitudes above 90000 feet to investigate thermal heating problems due to skin friction at high speeds.

It was also known as the thermal barrier, kinetic heating, thermal soaking or just soaking.

*See Aerodynamic heating.*



**Thermodynamics** Thermodynamics is the branch of science that deals with the conservation of energy involving the transfer of heat into other forms of energy and its relationship for example, with chemical and electrical energy and including all energy types within the universe. It deals with the availability of heat to do work and the direction of heat flow, from a hot source to a cooler source, and the changes in the internal energy of the system. Thermodynamics states all the individual particles of the system eventually resort to random motion of disorder. Because randomness (entropy) is continually increasing all energy in the universe is decreasing or 'slowly running down'.

Basic thermodynamics also appears in our study of aerodynamics, in particular, in compressible flows. It involves the relationship between temperature, pressure, and volume, which are all functions of entropy. An excellent example of the use of thermodynamics is in the study of airflow through jet engines, known as the Brayton cycle and in hypersonic flight.

The beginnings of thermodynamics (although not recognized as such) originated in 1662 when Sir Robert Boyle (1627–1691) an Irish physicist and chemist, published the first of his laws concerning the relationship between temperature and pressure. However, the histories of true classical thermodynamics dates back to the time of Nicolas Leonard Sadi Carnot (1796–1832) a French physicist and engineer; he is considered the Father of Modern Thermodynamics. In 1824, Nicolas Carnot introduced in his book what is now known as the 1<sup>st</sup> Law of Thermodynamics. Several other physicists were also responsible for discovering and introducing thermodynamics to the science community. In 1858, James Prescott Joule (1818–1889) an English physicist introduced the term thermodynamics to describe the scientific relationship between heat flow and temperature changes.

At the start of the 19<sup>th</sup> Century, the steam engine had been introduced as the main source of motive power for many applications, including the beginning of the railway systems. Improving the efficiency of the steam engine was of paramount

importance to the burgeoning Industrial Revolution in Britain from the second half of the 18<sup>th</sup> Century into the first half of the 19<sup>th</sup> Century. Philosopher of science Thomas Khun credits the 2<sup>nd</sup> Law of Thermodynamics initially to Nicolas Carnot. This law was further refined and described precisely in 1850–1 by Rudolph Clausius (1822–88) a German physicist. Clausius discovered that heat flowed from hot to cold (and not the reverse) and a fraction of the energy was dissipated and therefore, unable to do work. This is the basis of the 2<sup>nd</sup> Law of Thermodynamics.

Meanwhile in the United Kingdom William Thomson, Lord Kelvin (1824–1907) also introduced his version of the 2<sup>nd</sup> Law of Thermodynamics, between 1843 and 1848, in conjunction with James Prescott Joule (1818–1889). William George Rankine (1820–1880) a Scottish engineer and scientist, was also involved with thermodynamics and wrote the first textbook on this subject in 1859.

**See Entropy; Brayton cycle; First Law of Thermodynamics; Second Law of Thermodynamics.**

**Thickness** The term thickness refers to the maximum depth (or maximum thickness) between the upper and lower surfaces of a two-dimensional wing section, measured from the top and bottom outer skin surfaces. Maximum thickness and its distance measured from the leading edge, expressed as a percentage chord, defines the airfoil's shape.

- It is British convention to measure the thickness perpendicular to the chord line.
- It is American convention to measure the thickness perpendicular to the camber line.

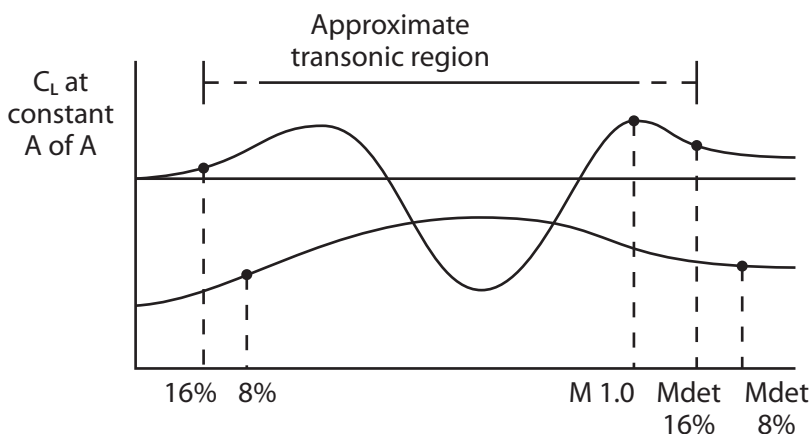
It is also known as thickness distribution.

**See Diagram 4, Airfoil Terminology; Maximum thickness.**

**Thickness/chord ratio** The thickness/chord ratio of a wing is a measure of the wing's thickness (depth) compared to its chord (width) expressed as a percentage or ratio.

The thickness/chord ratio of the wing is a compromise of requirements: thinner wings (low thickness/chord ratios) are required for high-speed flight and thicker wing are required for high lift, wing strength and fuel storage. The thickness measurements are taken at the airfoil station where the wing depth is greatest. A propeller blade's thickness/chord ratio is approximately 4% at the blade tips and increases to around 25% at the blade root.

Having less camber, the thin airfoil does not accelerate the airflow over the upper surface to such a relatively high-speed when the aircraft is flying at its cruise speed. Although the lift coefficient is lower than the thicker wing, the lift coefficient variation with speed is less and it peaks in the transonic region providing better longitudinal stability. It is the high-speed aircraft, which enjoy the advantages of low thickness/chord ratio wings, delaying the boundary layer separation and the onset of the transonic drag rise and thus raising the critical Mach number and the drag divergence Mach number.



**Diagram 86, Thickness/chord ratio & Lift Coefficient**

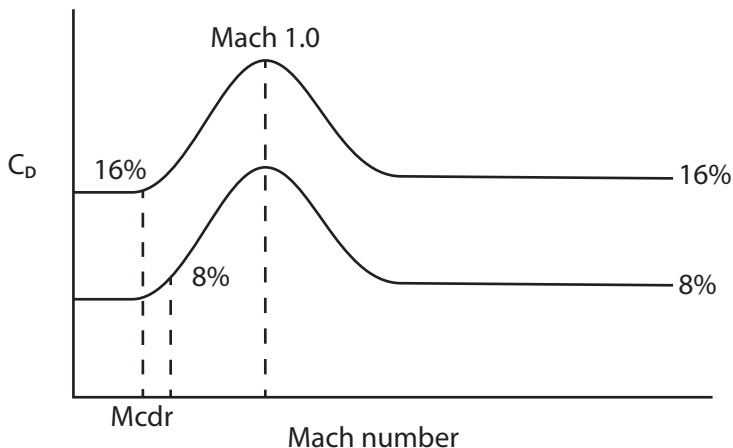
Flying in the transonic region, stability and control are substantially improved with low thickness/chord ratio wings. The disadvantage of low thickness/chord ratio wings is their relatively high landing speed due to the wing's lower maximum lift coefficient; this calls for high lift devices to reduce landing speeds and distance. Obviously, the reverse is required for slow flying subsonic aircraft, which benefit from higher thickness/chord ratio wings. **Diagram 86, Thickness/chord ratio & Lift Coefficient**, shows the variation in lift coefficient at constant angle of attack for two wings with 8% and 16% thickness/chord ratios.

The thickness/chord ratio can be classified under the headings of aerodynamic thickness/chord ratio and geometric thickness/chord ratio, as listed below:

- the aerodynamic thickness/chord ratio is the apparent thickness of the wing as the airflow sees it when flowing over a sweptback wing
- the geometric thickness/chord ratio is the actual physical measurement of the wing.

The above comments have considered the thickness/chord ratio to be constant across the wing's span. In reality, the thickness chord ratio varies from root to tip, with the greater thickness at the root. A greater thickness/chord ratio provides a greater increase in the flow's velocity, which can cause compression over the wing root, known as wing root compression. Weak shockwaves may form and merge into one normal shockwave over the rear of the wing's surface.

**Diagram 87, Thickness/Chord Ratio & Drag Coefficient**, shows two drag coefficient curves for a slim, 8% thickness/chord ratio wing, and a 16% thicker wing section. It can be seen the thin, 8% thickness/chord ratio wing has less drag and the drag rise due to compressibility occurs at a higher Mach number. The rise in the drag curve is attributed to the transonic increase of wave drag, increasing to a maximum at Mach 1.0. The values of 8% and 12% were derived from wind tunnel tests to compare a 1948 wing (16%) with a later wing from 1953 of 8%.



**Diagram 87, Thickness/Chord Ratio & Drag Coefficient**

Many aircraft of WWII era and other aircraft with high-lift wings have a thickness/chord ratio of approximately 14–18%, a relatively thick airfoil. General aviation aircraft have thickness/chord ratios of approximately 10–12%, while modern sweptback wing jet fighters have ratios down to 4–9%. For example, the thickness/chord ratio for the Lockheed F104 Starfighter's wing is a very slim 3.4%.

It is also known as the thickness ratio.

See *Diagram 4, Airfoil Terminology; Camber and camber line; Aerodynamic thickness/chord ratio; Drag divergence Mach number ( $M_{DIV}$ )*.

**Thin airfoil theory** The thin airfoil theory assumes a very thin, cambered wing section as being a single line shaped the same as the airfoil's mean camber line. The airflow over the airfoil is considered an inviscid, incompressible flow with a series of vortices located along the camber line. Aerodynamicists in their research work, favor the thin airfoil theory for calculating the air pressure, forces, and moments acting on thin airfoils, which to some extent can also be applied to airfoils with depth.

The lift coefficient curve shows how the lift varies linearly (when plotted against angle of attack) by approximately  $2\pi$  per radian change in angle of attack (0.1096/deg). Viscosity and thickness effects cancel out and are ignored in the calculations.

Dr. Maxwell Munk (1890–1986) further developed the thin airfoil theory in 1922, after the basic theory had been put forward independently by Professor Martin Kutta (1867–1944) in 1902 and Professor Nikolai Joukowski (1847–1921) in 1906. Theodore Theodorsen (1897–1978) of NACA improved the knowledge of thin airfoil theory.

*See* **Kutta-Joukowski theorem.**

**Thin supercritical wings** Thin supercritical wings are employed on many types of high-speed supersonic jet fighter aircraft. The thin wing is essential for high-speed flight. The Northrop/Grumman X-29A research aircraft was used to study the effects of thin supercritical wings. The X-29 is on display at the National Museum of the United States Air Force at Dayton, Ohio.

*See* **Supercritical airfoil; Thickness/chord ratio.**

**Thin wings** The very early aircraft up until the WW I era, had thin wings where the upper and lower surfaces were curved and almost parallel to each other, with little depth, (modern Flexiwing microlights and yachts also use thin wings). They work on the Bernoulli principle to produce lift on the upper surface, combined with the increased pressure on the curved under surface, to produce the required pressure gradient and curved downward flow.

The German aviation pioneer and engineer, Otto Lilienthal (1848–1896) in his book *Birdflight as the Basis of Aviation* in 1889, introduced this description.

**Three-dimension (3-D) vortex lift** *See* **Vortex lift.**

**Thrust** Thrust is the force imparted by a propeller or jet engine to provide the forward motion of the aircraft. It is generally measured as thrust horsepower when delivered by a propeller or thrust-pounds delivered from a jet engine. The SI unit is the Newton (N).

One pound of thrust is equal to one horsepower at a flight speed of 325 Knots (375 MPH) and thrust is less below this figure and greater above it.

The cruise thrust of a propeller-powered aircraft can be calculated from the formula:

$$\text{Cruise thrust (propeller)} = \text{PDA} \times (V + V_1) \times \rho \times V_o$$

Where PDA = propeller disc area, sq. meter or sq. ft.

V = aircraft velocity, meters per sec or FPS

V<sub>o</sub> = outflow velocity, meters per sec or FPS

V<sub>1</sub> = inflow velocity, meters per sec or FPS

ρ = air density kg/m<sup>3</sup> or lbs/cu.ft.

It should be noted, thrust and power is related; however, they both have their own individual definitions.

*See Diagram 60, Thrust & THP Required & Available Curves; Propwash-thrust.*

**Thrust axis** The flow direction of the thrust force acts along the thrust axis, which is aligned with the axis of propeller rotation or the jet engine's propelling nozzle.

**Thrust diamonds** *See Shock diamonds.*

**Thrust differential** The difference between thrust available and thrust required at a given power setting is known as the thrust differential.

*See Diagram 60, Thrust & THP Required & Available Curves.*

**Thrust/drag ratio** The thrust/drag ratio (T/D) compares the amount of thrust present compared to the aircraft's total drag. Better aircraft performance is achieved with a high thrust/drag ratio.

**Thrust equation** The thrust from a gas turbine engine (jet engine) depends on the mass times acceleration of the ejected gas flow through the exhaust, which produces a force to propel the aircraft forwards. This can be shown by the common equation for force:

$$\text{Force} = M \text{ times } a \quad (F = Ma)$$

Where  $F$  = force

$M$  = mass

$a$  = acceleration.

**Thrust face** A propeller's thrust face corresponds to the wing's lower surface, being relatively flat. It is that part of the propeller seen for example, by the pilot sitting in a single-engine aircraft.

See **Blade face**.

**Thrust horsepower available** The thrust horsepower available curve, as shown on *Diagram 60, Thrust & THP Required & Available Curves*, represents the thrust horsepower available from the propeller after allowing for the loss of brake horsepower due to the inefficiencies of the propeller. The thrust horsepower available varies and increases with the aircraft forward speed and is calculated for each speed from drag times velocity.

See *Diagram 60, Thrust & THP Required & Available Curves*.



**Thrust horsepower required** The thrust horsepower required is the power required to equal the aircraft's aerodynamic drag; the power required varies as the cube of the velocity. The thrust horsepower required curve is therefore, of a similar shape to the familiar total drag curve. Thrust horsepower required is calculated from the following formula:

$$\text{THP} = \frac{\text{Thrust} \times \text{Velocity}}{550}$$

Where Thrust = prop thrust-lb

Velocity = FPS, MPH or Knots

550 = factor for FPS

375 = factor for MPH

325 = factor for Knots

$$\text{or THP} = \frac{\text{drag} \times V (\text{FPS}) \times 60}{33000}$$

*See Diagram 60, Thrust & THP Required & Available Curves; Diagram 20, Total Drag Curves.*



**Photo:** The BAe 146 jet airliner has four Textron Lycoming LF 507 turbofans pylon mounted below the wings. The thrust line is very close to the aircraft's longitudinal axis – reducing the pitching forces with power changes.

**Thrust line** The engine's thrust line location, acting along the engine's axis, is one force that determines the aircraft's pitching characteristics with an increase or decrease of power. A high thrust line may pitch the nose down with power application and vice versa for low-slung engines such as those on Boeing jet transports.

*See Forces in cruise flight – airplane.*

**Thrust/power ratio** The ratio of thrust to power of a piston-engine is approximately 2.5 pounds force per horsepower.

**Thrust terminology** The propeller is considered to be 'free standing' when it is unaffected by the presence of the fuselage or engine nacelle behind the propeller. In this case, we refer to the thrust as 'free thrust'. However, if we take into consideration the presence of the fuselage or nacelle and its affect on the propeller slipstream, we then refer to the disturbed slipstream as 'apparent' or 'gross thrust'. Going one-step further, if we subtract the drag from the gross thrust (caused by the slipstream flowing over the fuselage or nacelle), we have 'propulsive thrust' or 'net thrust'. Propulsive thrust is always a constant fraction less than the apparent thrust, due to the drag being proportional to the slipstream velocity squared.

**Thrust/torque ratio** The greatest amount of propeller thrust for the least amount of engine torque determines the propeller's thrust/torque ratio.

*See Diagram 67, Propeller Forces in Cruise Flight.*

**Thrust/weight ratio** The ratio of thrust to weight provides an indication of the aircraft's performance characteristics. The greater the thrust/weight ratio, the better is the aircraft's performance, especially the rate of climb, due to greater excess thrust. Adding weight to the aircraft in the form of fuel, passengers or cargo, etc, will reduce the thrust/weight ratio and aircraft performance will reduce.

**Tilt-prop** *See Prop-rotor.*

**Tilt rotor system** In an effort to produce an aircraft capable of vertical take-off and landing (VTOL) and then transition to horizontal flight, the tilt rotor/prop idea was introduced with transverse rotors.



**Photo:** The Bell Aircraft XV-3A tilt rotor research aircraft is now located in the National Museum of the United States Air Force at Dayton, Ohio.

The tilt rotor system rotates the engine and propeller up to  $90^\circ$  from the horizontal to produce VTOL flight and then swivels down to the normal position for horizontal flight. The wing remains fixed in its normal location about which, the engines and propellers combined will rotate.

The Bell Aircraft XV-3A tilt rotor research aircraft was developed with a fuselage mounted engine driving two prop/rotors via drive shafts, mounted at the wingtips, which were capable of rotating through  $90^\circ$  to provide the required horizontal or vertical thrust. The wing remains fixed in its normal location about which the

engines and propellers rotate. The XV-3A's first flight was on First flight on 11 August 1955.

The Bell XV-15 tilt rotor was a twin-engine research aircraft designed with an engine at each wingtip driving its own propeller. The engine/prop combination rotated as one unit to provide VTOL and forward flight.

It is also known as a prop-rotor.

**Tilt wing** The tilt wing principle is a variation on the tilt rotor concept. The Chance Vought/LTV XC-142A tilt wing research aircraft is but one example developed to test the feasibility of tilting the complete main wing 90° up from the horizontal with the propeller and engines attached to vary the direction of propeller thrust.



**Photo:** The Chance Vought/LTV XC-142A tilt wing aircraft is now on display in the National Museum of the United States Air Force at Dayton, Ohio.

The four engines installed are General Electric T64-GE-1 turboprops, rated at 2850 shp each. Its maximum all up weight for (VTOL) operations was 42500 lb (19278 kg) and 44500 lb (20638 kg) for normal operations. It made its first flight on 29 September 1964 and its first VTOL transition to horizontal flight and back to hover on 11 January 1965.

**Tip chord** The tip chord is the distance measured from the leading edge to the trailing edge of the wingtip in a line parallel to the OX longitudinal axis.

**Tip droop** Tip droop is used on a few supersonic aircraft to alter the aerodynamic characteristics of the wings.

Compression lift is induced at supersonic speed in the channel formed by the wings under surface, the drooped tips, and the fuselage. This has the effect of moving the aerodynamic centre forward and reducing trim drag.

**Tip divergence/loss** The tips of a wing and especially sweptback wingtips may experience a loss in lift due to the spanwise component of the airflow over the wing. The loss of wing tip lift results in a loss in lateral stability due to tip divergence.

Propeller blade tips are inherently inefficient due to the very high tip speed close to, or exceeding the speed of sound resulting in a loss in thrust from the propeller tip area.

**Tiperons** Ailerons combined with tip flaps are known as tiperons. They are used mainly on flight vehicles such as missiles, rockets, drones and a few specialized aircraft.

**Tip-path plane** The tip-path plane is the circular path traveled by the propeller blades or helicopter rotor blades. It is parallel to the plane of rotation and normal to the axis of rotation. It can be considered as a fixed plane of rotation for a propeller but can

be tilted, within limits on a helicopter's main rotor system by the pilot's cyclic control for directional control. The tip-path radius reduces with increased coning angle; the aircraft's forward speed is ignored.

It is also known simply as the tip path.

*Compare with* **Plane of rotation.**

**Tipsail** Another name for a winglet is a tipsail, but the term is less common. The term was introduced by Beechcraft in 1983, for the winglets mounted on their futuristic Starship 1 twin-turboprop, executive aircraft. Mounted on the tips of the main wings, they also doubled as vertical stabilizers.

*See* **Wingtip devices.**



**Photo:** Beechcraft Starship NC-23 at the Pima Air and Space Museum, Tucson, Arizona.

**Tip speed** The propeller's tip speed is represented by vector A-B the tangential velocity on **Diagram 68, Propeller Pitch**. The combined forward velocity of the aircraft and propeller is ignored.

A high propeller tip speed can produce a distinctive harsh sound if the blade tips reach supersonic speed and develop shockwaves– a frequent occurrence on piston-engine aircraft with constant speed propellers during take-off. Likewise, a helicopter's high rotor blade tip speed can cause vibration, noise and increased drag.

A helicopter's rotor blade system operating in the almost horizontal plane will have varying blade tip speeds due to the machine's forward motion – increased tip speed on the advancing blade and reduced tip speed on the retreating blade. The tip speed of the propeller is determined by the blade length and RPM only.

**Tip speed ratio** The tip speed ratio is the ratio of the propeller, or rotor blade's tip speed to the aircraft's forward motion.

**Tip stall** A wingtip that stalls before the wing root is an undesirable situation, a degradation of lateral stability with the possibility of a spin developing, reduced aileron effectiveness and a possible loss of control in the rolling plane. Wingtips with relatively high lift coefficients may stall before the wing root stalls. Many light aircraft have a stall strip mounted on the inboard leading edge to trip the wing root to stall first, thus preventing a tip stall. Sweptback wing jet transport aircraft may have a fence to prevent the spanwise boundary layer from flowing towards the wingtips to prevent a change in the lift distribution, which increases the tip lift coefficient and decreases the energy in the boundary layer, result in wingtip stalling.

See **Tip divergence/loss**.



**Torque force** Torque is the turning moment of a force or couple.

Torque is a force or a combination of forces acting in a circular motion about an axis. It can be defined as the force times radius, known as a lever arm. An increase in either force or radius results in an increase in torque. The lever arm is the distance from the axis of rotation (pivot) to the position of the applied force, or to put it simply for example, the length of a spanner. The greater the lever arm, the greater is the torque. Therefore, torque is proportional to the lever arm. Likewise, the greater the applied force, the greater is the angular acceleration and torque.

Newton's third law of motion states, 'for every action there is an equal and opposite reaction'. In providing power to turn a propeller for example, the engine produces a torque component (engine torque), which is a force acting in the same plane and direction as the propeller rotation. At a constant power setting, the engine torque is balanced by the equal and opposite force of propeller torque, in accordance with Newton's third law. On an aircraft with a fixed-pitch propeller, the engine RPM will remain constant as long as these two forces remain in balance. A change of power setting or aircraft speed will change the value of the engine torque or propeller torque respectively. This will result in a change in engine RPM.

In SI units, torque is measured in Newton-meters (N-m).

*See **Moment; Propeller torque force; Torque roll and stall.***

**Torque roll and stall** The engine torque will produce an equal and opposite torque reaction on the propeller. This will create a turning moment, which will tend to rotate the aircraft around its OX longitudinal axis in the opposite direction to the propeller's rotation. With a 'right-handed' propeller (one rotating clockwise when viewed from the rear), it will cause the aircraft to rotate or roll to the left, in compliance with Newton's third law of equal and opposite reaction.

Some aircraft types – piston-engine fighters of WW II are a good example – with a high power to weight ratio, can develop a torque

roll during take-off or in a go-around situation. The up-going wing meets the relative airflow at a reduced angle of attack while the down-going wing meets the airflow at a greater angle of attack, which if it exceeds the critical angle, a torque stall develops with a resulting total loss of control.

*See Propeller torque force; Torque force.*

**Torsional aileron flutter** A wing pitching around its OY lateral axis will cause its leading edge to oscillate up and down around the wing's torsional axis. This can lead to aileron reversal.

The main wing and ailerons have greater aeroelasticity than the shorter tailplane and is therefore, more likely to suffer from torsional flutter, although it is not unknown for the tailplane to suffer the effects of flutter first.

Torsional flutter can be reduced by installing a mass balance, which moves the centre of gravity closer to the hinge line. Hydraulic actuated controls that are common on all large aircraft are less susceptible to flutter due to their inherent damping in the hydraulic system. Mass balance is used on smaller aircraft, which are more likely to suffer from torsional aileron flutter.

*See Wing divergence – aeroelastic; Flexural aileron flutter; Aileron reversal.*

**Total aerodynamic reaction force** *See Total reaction.*

**Total drag** An aircraft's total drag is divided into two basic groups of drag – drag due to lift and parasite drag. Parasite drag can be further divided into the sub-groups of profile drag (form drag and skin friction) plus body drag and extra drag.

*See Drag.*

**Total entropy** *See Entropy.*

**Total head** See **Total pressure**.

**Total lift** The total lift is the vertical component of the resultant force due to the aerodynamic reaction acting on the aircraft's wings. In normal straight and level flight, the total lift is equal and opposite of aircraft weight.

**Total pressure** The total pressure is the pressure of a fluid (air) moving over an aircraft in flight. In an incompressible flow, it is the total sum of the dynamic and static pressure.

The total pressure formula below represents Bernoulli's equation for an incompressible flow:

$$P = p + \frac{1}{2} \rho V^2$$

Where  $P$  = total pressure

$p$  = static pressure

$\frac{1}{2}$  = a constant

$\rho$  = air density

$V^2$  = velocity.

The total pressure is also known as the total head, pitot, or stagnation pressure. It can be considered to have the same meaning as impact pressure or pitot pressure.

See **Dynamic pressure**; **Static air pressure**; **Stagnation pressure**.

**Total reaction** With the wing or propeller advancing through the air there is an upwash of air in front of the airfoil and a downwash from the trailing edge. The net result of this upwash and downwash is a general downwash of the relative airflow over the airfoil. Because the total reaction is at a right angle to the relative airflow, it will be tilted rearwards from the vertical relative to the airfoil or propeller blade element.

The vertical component of the total reaction is the lift force acting at right angles to the free stream airflow. The horizontal component

of the total reaction acting parallel to the free stream airflow represents the induced drag, or to use the modern terminology, trailing vortex drag.

The diagram for a propeller or rotor blade's total reaction shows the total reaction can be sub-divided into two sets of vectors. As mentioned above, one set of vectors contains the lift and aerodynamic drag forces, while the other set contains the thrust and propeller/rotor blade torque forces.

It is also known as the resultant force or the total aerodynamic reaction force.

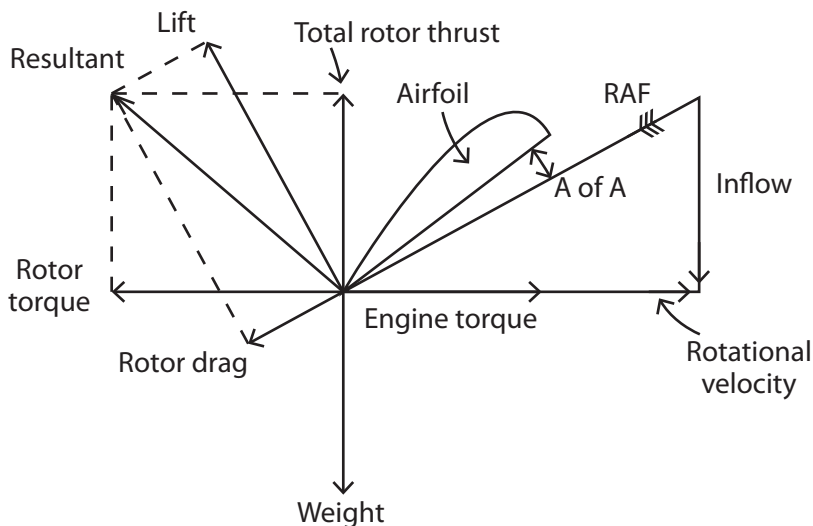
*See Diagram 67, Propeller Forces in Cruise Flight.*

**Total rotor thrust** The total rotor thrust is the sum of the rotor thrust forces from both or all rotor blades on a helicopter's rotor disc.

Some pilots find the definition of the total rotor thrust confusing and avoid using the term. Is it the same force as lift, or not? No, well not exactly! An explanation is required. When looking at a diagram of a helicopter's rotor disc forces, the vertical arrow at right angles to the plane of rotation and pointing straight up or nearly so, is usually labeled as the lift, which true to say, it is. The lift from the rotor blades equals the helicopter's weight. However, relating the total rotor thrust to this force can be the cause of confusion.

The rotor blade produce an aerodynamic reaction known as the resultant force, or more commonly the total reaction. The total reaction (resultant) is tilted rearwards several degrees from the axis of rotation, the same as found on an aircraft wing (assume the rotor blade is moving from left to right on **Diagram 88, Total Rotor Thrust**. The resultant can be divided into two pairs of associated vectors of lift and rotor blade drag and the vectors of total rotor thrust and rotor torque. Rotor blade drag is parallel and downstream of the relative airflow vector. The lift vector is at right angles to the relative airflow (RAF); note this position. Rotor torque is equal and opposite of engine torque, which is in line

with the plane of rotation and the total rotor thrust is therefore, at right angles to it. The total rotor thrust is equal and opposite to the helicopter's weight.



**Diagram 88, Total Rotor Thrust**

Our conclusion to this dilemma is the following; The contribution of each rotor blades' individual lift (rotor thrust) is tilted inwards towards the axis of rotation, due to the rotor blade's coning angle, to produce the total rotor thrust vector. The total lift force from both or all rotor blades is at right angles to the plane of rotation. Therefore, the total lift from both or all of the rotor blades is the same force as the total rotor thrust. **Diagram 88, Total Rotor Thrust**, shows all the forces acting on one rotor blade. It is the inflow vector, which causes the confusion in the terminology. To simplify the whole concept, think of it this way; the term total rotor thrust is the *name of the force* and lift is *what it does*!

Compare with **Diagram 67, Propeller Forces in Cruise Flight**.

**Total rotor thrust/rotor torque ratio** Considering a helicopter's rotor disc, the total rotor thrust (TRT)/rotor torque ratio is akin to an airfoil's lift/drag ratio.

When the rotor disc is operating at its maximum TRT/torque ratio the rotor blades are at a small angle of attack to the relative airflow, which produces the maximum lift for the least possible rotor drag.

*Compare with **Diagram 88, Total Rotor Thrust.***

**Total surface area** The total surface area of the wing includes the plan view of both upper and lower surfaces combined.

*Compare with **Wing area.***

**Townend ring** A Townend ring is a short-chord airfoil section surrounding a radial engine.

It is designed to reduce drag from the engine cylinders and improve the cooling airflow over the engine by deflecting the air inwards. A speed gain of 10–15 MPH was claimed due to the drag reduction; the drag without the Townend ring was about four times greater! The Townend ring was 'invented' in 1929, by Dr. H. C. H. Townend at the British National Physical Laboratory.

The Townend Ring was used successfully on the British built Gloster Gauntlet, Vickers Wellesley, and the Westland Wallace.

The previous year, Fred Weicke (1899–1993) a NACA aerodynamicist had introduced the long-chord NACA cowling. It was claimed the Townend ring was not as efficient as the American NACA version. However, commercial pilots praised it but fighter pilots were averse to it, claiming it reduced their forward view between the cylinders, an important consideration during combat flying.

*See **NACA cowling.***

**Tractor propellers** Tractor propellers are mounted on the ‘nose’ of the aircraft or on the leading edge of the wing. They pull the aircraft through the air as opposed to pusher propellers that push from the rear of the engine.

Ambroise Goupy (1876–1951) a lesser-known early aviation pioneer designed and built the first tractor propeller biplane, the Goupy II circa 1908. With the assistance of Louis Bleriot (1872–1936), the Goupy II used a Bleriot fuselage converted from pusher configuration to tractor propeller layout.

Leon Levavassour (1863–1922) was a French powerplant engineer, aircraft designer, and inventor (he invented the V-8 engine). He designed the Antoinette IV monoplane in 1908, as a tractor prop aircraft; this became the most commonly accepted layout for planes that were to follow, as opposed to the pusher propeller layout of the earlier aircraft. In conjunction with Louis Bleriot, Levavassour was instrumental in establishing the tractor prop layout for piston/prop aircraft.

*See **Propeller stability**.*

**Trailing edge flaps** As the name suggests, the flaps are mounted on the trailing edge of the main wing to provide extra lift and drag during slow-speed flight.

*See **Flaps**.*

**Trailing edge suction** An area of low air pressure, or suction occurs near the extended trailing edge flaps. The trailing edge suction therefore, plays its part in helping the airflow to follow around the wing’s leading edge. The deployed flaps create greater camber over the wing’s upper surface inducing the air to flow faster and thus delaying separation and increasing the lift. Therefore, with flaps deployed the stalling speed is reduced.

**Trailing vortex drag** An alternate name for induced drag or lift-dependant drag.

**Trailing vortex sheet** The two airflows over the upper and lower wing surfaces meet at the trailing edge at a rake angle forming a trailing edge vortex sheet. It induces a downwash over the wing, causing induced drag. The trailing vortex sheet off each wing rolls up into and combines with, the wingtip vortex to form one large-radius vortex some distance aft of each wing. The trailing vortex sheet commences as the starting vortex and when combined with the wingtip vortex it forms the horseshoe vortex system.

*See Induced drag; Geometric angle of attack; Effective angle of attack.*

**Trailing vortices** *See Wingtip vortices.*

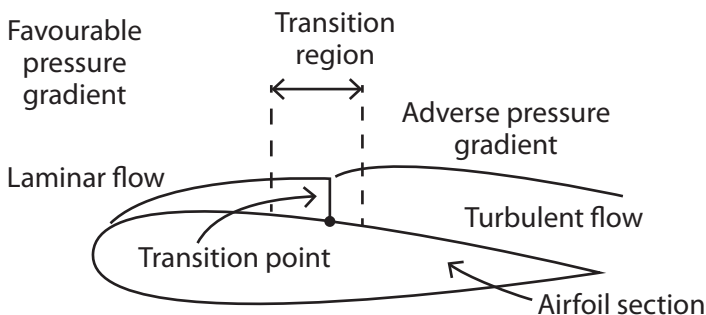
**Transition point/region** The transition point/region is the location on the wing's upper surface where a laminar boundary layer flow changes to a turbulent flow.

The exact position of the transition point is difficult to determine with accuracy and therefore, a zone known as the transition region is accepted as being more descriptive. More than one transition region may develop on the wing's surface and amalgamate into one large turbulent boundary layer on the aft portion of the wing.

The transition point's location depends on the adverse pressure gradient, air speed, surface roughness, wing sweep, and Reynolds number. The point of least pressure determines the transition point but air speed has a greater influence where an increase in air speed (and thus Reynolds number) moves the transition point forward. The transition from laminar to turbulent flow in the boundary layer is dependent on the Reynolds number being sufficiently high.

The wing's surface condition has a substantial effect on the location of the transition point. Surface roughness in the form of rounded rivet heads, imperfection on the skin surface, raindrops





**Diagram 89, Transition Point & Region**

on laminar flow wings and icing all reduce the kinetic energy of the boundary layer and thus the airflow velocity. This reduces the wing's lift and the maximum lift coefficient. Icing, particularly over the leading edge and the first 20% MAC of the wing's top surface will cause the transition point to move forward. Aft of the transition point, a greater area of the wing is affected by the thickening, turbulent flow and hence produces greater skin friction with increasing air speed.

Reference to **Diagram 89, Transition Point & Region** shows a viscous flow in the boundary layer, which is initially a smooth laminar flow gradually thickening to the start of the transition region where it increases in depth and becomes a turbulent flow. The airflow outside of the boundary layer remains an inviscid flow. Professor Ludwig Prandtl (1880–1953) and his co-workers at the Gottingen Aeronautical Research Centre in Germany, investigated as early as 1904, the stability of the boundary layer with its transition point and region and its adverse effect on aerodynamic drag.

Also known as the laminar transition point.

See **Separation point & flow.**

**Transition spin** Incorrect control input during the recovery from a normal spin can place the aircraft into a transition spin, which develops into an inverted spin.

The transition spin is caused by trying to lower the nose too quickly – too much down elevator – which shields the fin/rudder of essential airflow resulting in the yaw becoming a roll. The rate of spin accelerates (like a ballerina doing a pirouette) and the aircraft attitude passes through the vertical to become an inverted spin. With anti-spin rudder applied, this becomes pro-spin when the aircraft is inverted; instead of recovering, the spin continues.

*See Spinning.*

**Translating tendency** A helicopter's translating tendency is due to the torque couple acting on the fuselage due to the rotation of the main rotor disc plus the addition of the tail rotor force causing the helicopter to drift to the right in no-wind conditions, assuming the rotor blades turn anti-clockwise when viewed from above.

*See Tail rotor drift.*

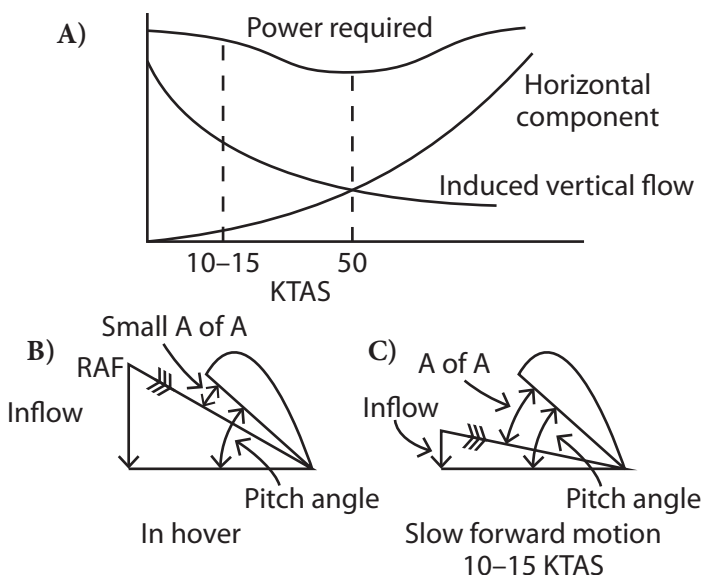
**Translational flight** An aircraft in horizontal flight with lift generated by the aircraft's wings, as in straight and level translational flight. For a helicopter, it is the change in flight condition from the hover to forward flight.

**Translational lift** When the helicopter is in the hover, the induced inflow is vertical and considerable power is required to produce it.

*Diagram 90, Translational Lift*, shows when moving forward from the hover with increasing speed, the vertical induced inflow component reduces. The resultant of these two vectors (the relative airflow approaches the rotor blade from a lower angle increasing the angle of attack as the rotor disc tilts forward with increasing

helicopter forward speed, which increases the forward inflow velocity.

At a translational forward velocity of 10–15 Knots, a greater mass of air through the rotor disc improves the aerodynamic efficiency. Because the angle of the induced inflow decreases, the rotor blade's angle of attack increases. With an increasing inflow velocity due to the increasing forward speed, there is an increase in total rotor thrust and less power and collective (reduced angle of attack) is required than in the hover. This is the translational lift or simply, translation.



**Diagram 90, Translational Lift**

From another point of view, less power is required during translation due to the rotor's total reaction vector becoming more aligned with the axis of rotation. Therefore, the rotor drag is reduced and so less power is required to maintain lift, or with constant power, the helicopter's rate of climb is increased. The greater the disc tilt, the larger is the component adding to the induced flow. As the forward speed increases further to around 50

Knots, the inflow angle and rotor blade angle of attack both reduce, which requires an increase of power to compensate. The power will then be greater than the power required for translational lift but less than the power required for the hover. **Diagram 90, Translational Lift** shows this. Translational lift is also present, to a varying degree, when hovering in a wind.

**Transonic area rule** It is more commonly known simply, as area rule or Whitcombe area rule.

**Transonic airflow** Transonic airflow is by convention, the velocity of the airflow at any speed between Mach 0.8 and Mach 1.2.

When the aircraft accelerates, transonic flow first appears on top of the wing due to its camber and other parts of the aircraft such as over the cockpit canopy. Therefore, some part of the flow will be supersonic while other parts will be subsonic. Weak shockwaves (sonic lines) will appear in the supersonic parts of the airflow, with the flow reducing to a speed below Mach 1.0 behind the shockwave.

Aircraft designed for transonic or supersonic flight have a sharp nose, a very streamlined fuselage, and thin wings. Therefore, the transonic flow over the aircraft is reached at a cruise speed close to Mach 1.0. Aircraft with less streamlined proportions and deeper thickness wings for example, will experience Mach 1.0 airflow at a lower cruise speed.

Between the values of approximately Mach 0.9 to Mach 1.2, the increase in the drag rise is too great for fuel-efficient cruise by any aircraft. Supersonic aircraft must continue to accelerate through this speed region to at least Mach 1.3 or higher for economical cruising. Shockwaves start to make their presence known at the plane's critical Mach number of around Mach 0.8. When the aircraft's speed has reached Mach 1.2 the airflow will by then, be re-stabilized with the lift coefficient reducing to a value below that of subsonic speed flight. This marks the lower and upper limits of the transonic region.

The Navier-Stokes equations are not applicable in the transonic region.

The term 'transonic' was coined by NACA engineers, Hugh Latimer Dryden (1898–1981) and John Stack, circa 1941 during their research into the transonic speed range.

*See* **Mach wave/wavelet; Shockwaves; Critical Mach number.**

**Transonic compressibility drag** An alternate term for wave drag.

*See* **Wave drag.**

**Transonic drag hump** The transonic drag hump is the result of a shockwave forming on the wing's upper surface. It is due to the increase of parasite drag caused by wave drag and the formation of boundary layer separation drag, which initially increases and then decreases with increasing Mach number through the transonic region to supersonic speeds. The transonic drag hump is shown on *Diagram 94, Wave Drag*, as a bulge on the new parasite drag curve.

*See* **Wave drag; Diagram 94, Wave Drag.**

**Transonic tunnel** A wind tunnel designed to operate with flow-speeds in the transonic speed range of Mach 0.8 to Mach 1.2.



**Photo:** The Lockheed F-22 Raptor fifth generation, supermanoeuvrability fighter aircraft has a trapezoidal main wing and twin tails. This F-22 Raptor is now on display at the National Air & Space Museum in Dayton, Ohio.

**Transverse axis** The OY transverse or lateral axes run parallel to a line from wingtip to wingtip through the centre of gravity. The aircraft rotates in pitch about the transverse or lateral axis.

It is also known as the transverse lateral axis.

**Transverse flow effect** The transverse flow effect describes the changes in airflow over and through the helicopter's main rotor disc in translational flight.

*See Translational lift.*

**Transverse lateral axis** *See Transverse axis.*

**Transverse roll** *See Inflow roll.*

**Trapezoidal wing** A trapezoidal wing is diamond shape in plan view having a sweptback leading edge and a forward sweep trailing edge. This shape produces a short span, low aspect ratio wing with a thin airfoil section; drag at high speeds is relatively less than that of a more conventional wing of similar size and the ride is smoother in turbulence at high subsonic speeds due to the higher wing loading. One disadvantage is the increased take-off and landing speeds and distance due to the high wing loading.

Trapezoidal wings are used on the Douglas X-3 Stiletto research aircraft, Lockheed F-104 Starfighter, and also the modern Lockheed Martin F-35 Lightning II and the Lockheed Martin/Boeing F-22 Raptor, to name just a few.

**Trim** With the tailplane/elevator set to counteract any undesirable pitching moments, the control stick force is relieved with elevator trim. The aircraft is then said to be 'in trim' when the sum of all pitching moments about the centre of gravity are zero and the aircraft is in a state of balance, or equilibrium. In straight and level flight, there exists only one angle of attack and air speed for any given weight where the aircraft is in equilibrium. That is to say,

the tailplane moments are equal and opposite to the main wing moments. A deflection of the trim tab will hold the control surface in that position relative to the wing to which it is attached.

To 'trim' the aircraft is the act of using the trim gear, either manually or via the electric autotrim.

**Trim tab** Trim tabs are hinged to the trailing edge of a primary control surface. Their purpose is to relieve the pilot from having to apply any long-term control pressure due to changing loads on the primary control surface. Cables from the cockpit trim wheel operate it, or electric servomotors are used on larger aircraft.

Trim tabs work in the opposite direction to the required movement of the primary control surface to which they are attached. To raise the elevators, for example, the trim tab hinges downwards, its lift then raises the elevator. All elevators have trim tabs of one type or another, but only larger aircraft have them included on the rudder and ailerons.

*See* **Trimming devices.**

**Trim drag** Trim drag is produced whenever a trim tab is deflected into the air stream and is no longer in-stream with its control surface. A forward centre of gravity loading is balanced by an up-elevator or elevon, which in turn, is required to be held in that position by the trim tab. In addition, not being streamlined with the airflow, the amount of drag can be quite considerable.

To reduce trim drag, some aircraft use variable incidence tailplanes set to the in-flight angle of incidence for the best streamline position and hence, least drag. Supersonic aircraft have their centre of pressure move aft with increasing speed, thus increasing the distance between the centre of pressure and centre of gravity, which results in a greater nose down pitching moment. The increased moment requires greater down elevon resulting in more trim drag. The cure on Concorde for example, is to transfer fuel aft to a holding/trim tank to maintain a constant centre of





**Photo:** The Chance Vought F4U Corsair has a trim balance tab on the port (left-hand) elevator and a trimming tab on the starboard (right-hand) side to reduce pilot input forces. This aircraft is located at the Pima Air & Space Museum, Tucson, Ar.

gravity to centre of pressure distance, limiting the nose down moment and trim drag. The second advantage in down elevons is they act as flaps.

**Trimming devices** Trimming devices (or trimmers) are small secondary airfoils attached to the trailing edge of a primary control surface. Briefly, they are divided into three basic groups as follows:

- trim tabs as the name suggests, are used to adjust the balance of the primary control surface to which it is attached to maintain the required aircraft attitude.
- servo tab, balance, and spring tabs are used to *lighten* the pounds of pull required from the pilot to move the control surface.
- Anti-balance tabs or anti-servo tabs are used to provide *heavier* control feel.

Frequent changes in the aircraft's speed, attitude, power settings, undercarriage position, freight, fuel or passenger loadings all change the centre of gravity and balance of the aircraft requiring adjustments on the primary flight controls. Without trimming devices, the pilot would be constantly required to hold the controls in a set position; this would be very tiring after a short time, hence the need for trimmers. Just about all light aircraft have at the very least, an elevator trim tab. Moving up to larger aircraft, they have trimming devices on the elevators, ailerons, and rudder. Trim tabs may be a permanently fixed ground adjustable type, pilot operated type or attached mechanically to the control surface.

The mnemonic to remember the different basic trim types is: THIS = tabs, horn, inset, and servo.

See **Trim tab; Balance tab; Horn balance; Inset hinge; Spring tab.**

**Trim point** The position on *Diagram 61, Pitching Moments V. Angle of Attack*, where the pitching moments are zero along the aircraft's stable slope, is known as the trim point.

See *Diagram 61, Pitching Moments V. Angle of Attack; Diagram 83, Tailplane & Trim Point*.

**Triple fin** The Lockheed Constellation piston-engine airliner is distinctive with its triple tail. Using three fins of lower height enabled the aircraft to enter hangars but still maintain maximum fin area for stability purposes.



**Photo:** The distinctive triple fin tailplane identifies the Lockheed Constellation, located in the Pima Air & Space Museum, Tucson, Arizona.

**Triple-slotted flaps** These are usually of the multi-vane Fowler type flaps found on large, jet transport aircraft.

The first flap vane closest to the wing's trailing edge is the foreflap. The second vane is known as the mid-flap or flap airfoil

and the final flap is called the trailing edge vane. The air cascades through the gaps between each vane re-energizing the airflow over the following flap.

Some aircraft may have only two vanes, known obviously, as double-slotted flaps. The early models of the Boeing 747 had triple-slotted flaps, which was changed to double-slotted flaps with current models.

*See **Flaps; Fowler flaps.***

**Trousers** This is now an archaic term used for wheel fairings, which streamline the airflow over the wheels on fixed-undercarriage aircraft to reduce the form drag.

They are also known as wheel pants.



**Photo:** The McDonnell Douglas C-17 Globemaster's T-tail reaches 55 feet 1 inch (16.79m) above the ground.

**Tsein H.** Hsue-shen Tsein (1911–2009) was one of the founding members of the Jet Propulsion Laboratory at Caltech in California. He was considered the ‘Father of Chinese Rocketry’ for his extensive work in rocket power research.

Hsue-shen Tsein with Professor Theodore Von Karman (1881–1963) developed the Karman-Tsein theory in 1939–1941. Their theory explained the effects of compressibility and the variation of the centre of pressure on airfoils with increasing Mach number.

**T-tail** A T-tail is a horizontal tailplane mounted on top of the vertical fin.

**Tufting** An aircraft undergoing flight-testing may have tufts of wool or string attached to parts of the airframe, which are to be studied for laminar or turbulent airflow. The tufts will show the condition of the airflow boundary layer where they lie.

**Turbulators** *See Vortex generators.*

**Turbulent flow** A thin laminar flow stays in contact with the forward part of the wing’s surface due to the favourable pressure gradient. The airflow gradually changes from a laminar to turbulent airflow in the transition region due to the greater shearing stress (skin friction) of the flow.

The turbulent boundary layer has greater energy and grows faster (thickens) and reaches a critical size of about 2 centimeters (1 inch) and tends to stay attached to the surface more readily beyond the transition region. Initially, the turbulent flow adheres to the wing surface better than a laminar flow – hence the success of vortex generators. However, it eventually becomes unstable and has difficulty in remaining attached resulting in a substantial increase in drag. The flow then breaks away forming eddies within the random motion of the flow with a further increase in turbulent flow and a possible forward flow, known as recirculation. This area is then known as the dead air region or the wing’s third region.



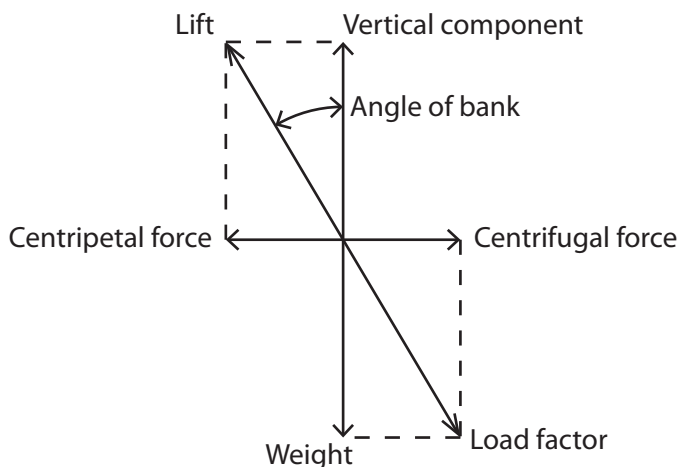
**Photo:** The British built Hunting H.126 research aircraft with tufting attached to its T-tail fin. The aircraft is on view at the RAF Museum Cosford, UK.

Therefore, it is of great advantage to retain an attached airflow over the wing for as long as possible. This will ensure that within the turbulent boundary layer a higher level of kinetic energy is retained. The effects of the adverse pressure gradient are delayed for a longer chord length, in turn reducing boundary layer separation and forward movement of the separation point. This reduces the drag increase and increases the maximum lift coefficient with increasing angle of attack.

The airflow over and below the wing meet at an angle known as the rake angle at the trailing edge, which causes flow interference resulting in a further thickening of the boundary layer and a reduced velocity. This is known as vortex wake drag.

**See Laminar flow airfoils; Boundary layer; Laminar boundary layer; Recirculation.**

**Turning** In order to change the heading of an aircraft in flight a force is required, which is provided by the wing's lift force being inclined in the direction of turn.



**Diagram 91, Turning Forces**

A turning aircraft is not in equilibrium even though all forces are in balance. It is turning and accelerating towards the centre of the turn with a constant forward speed. Reference to **Diagram 91, Turning Forces**, shows the lift force inclined towards the centre of the turn. The lift is divided into the vertical component to oppose the aircraft weight (mass) and a horizontal component to provide the necessary centripetal force (CPF) to turn the aircraft. The centripetal force, which obeys Newton's 2<sup>nd</sup> Law of Motion, is equal and opposite of the centrifugal force (CFF), which obeys Newton's 3<sup>rd</sup> Law of Motion. The centrifugal force is the force we feel for example, when riding in a car around a corner. The inclined, and increased lift force is opposed by the load factor, or the apparent increased weight (mass) which we could call the effective weight.

The centripetal force is the sideways force required to turn the aircraft, is found from the following formula:

Centripetal Force = weight times tan angle of bank

Given angle of bank =  $\tan 40^\circ$  (0.8390)

Aircraft weight = 4500 kg

Centripetal force = 4500 times 0.8390 = 3780 kg

The increased total lift component force can be found from the following formula:

Total lift = weight/cos angle of bank

Given angle of bank =  $\cos 40^\circ$

Aircraft weight = 4500 kg

Total lift =  $4500/(\cos 40^\circ) =$

$4500/0.7660 = 5874 \text{ kg}$

The load factor (effective weight) is the ratio of lift to weight and is found from the following formula:

Load factor = total lift/weight

Given total lift = 5874 kg

Aircraft weight = 4500 kg

Load factor =  $5874/4500 = 1.3$

See acceleration in a turn, angular velocity, centripetal and centrifugal forces, rate of turn, radius of turn, diameter of turn and turning.



**Twin fins** Twin fins mounted on jet fighter aircraft first appeared in 1967 with the introduction of the MiG-25. Unlike vertical twin tails, twin fins are canted outboard to improve direction stability and to maintain smooth airflow over at least one fin during yaw movements. Aeroelasticity is improved and they are more efficient at high angle of attack manoeuvres compared to a single tail.



**Photo:** The McDonnell F-18A Hornet jet fighter has twin fins canted outboard to improve directional stability.

**Twin tail** A twin tail or H-tail as the name implies, has two vertical fin/rudders at the end of the horizontal tailplane. A piston powered twin-engine aircraft with a twin tail has both fin/rudders lying within the propwash of each engine for improved low speed rudder authority.



**Photo:** The twin tails of the Beech S18D each have a smaller surface area than a single tail fin and have the advantage of always being immersed within the propwash. This aircraft is located in the Pima Air & Space Museum in Tucson, AR.

**Two-dimensional (2-D) inlets** Two-dimensional jet engine intake inlets are rectangular and are found on supersonic aircraft. A flat inlet ramp controls the airflow to the engine reducing the velocity of the approaching supersonic flow down to a subsonic flow, which the jet engine requires for the combustion process. With changing Mach number, the inlet plate angle adjusts to control the flow and to ensure maximum pressure recovery throughout the aircraft's speed range.



**Photo:** The Panavia Tornado's rectangular-shaped 2-D inlet is clearly shown in this front-on view. This aircraft is on display at the RAF Museum Cosford, UK.

The North American RA-5 Vigilante was the first aircraft to use 2-D inlets. This was followed by the BAC Concorde, MiG-25 Foxbat, Grumman F-14 Tomcat, Lockheed Martin F-22 Raptor, and McDonnell Douglas F-15 Eagle, which are all excellent examples of aircraft with 2-D inlets.

**Two & three-dimensional wings** A two-dimensional (or infinite) wing may be considered a wing with infinite span joining both wind tunnel walls, and devoid of any wingtip effects or vortex flow effect, which is found on real-time, three-dimensional wings.

A three-dimensional wing has a finite length as found mounted on an aircraft. Due to wingtip losses, the total lift and lift coefficient ( $C_L$ ) will be less than an equivalent two-dimensional wing. Assume for a constant chord wing a maximum lift coefficient equal to  $C_{L \text{ max}}$  times 0.93. For a tapered wing, assume a maximum lift coefficient equal to  $C_{L \text{ max}}$  times 0.95.

Note, for a three-dimensional wing, the term  $C_{L \text{ max}}$ , has a capital 'L' and for a two-dimensional wing the term uses a lower case 'l' and is known as the section lift coefficient.

The lift formula for a two-dimensional wing is slightly different to the commonly found lift formula. The equation for a two-dimensional wing is:

$$\text{Lift} = \pi \rho V^2 A \sin \theta$$

Where  $\pi = 3.142$

$\rho =$  air density

$V^2 =$  air speed

$A \sin \theta =$  angle of downwash.

Professor Martin Kutta (1867–1944) and Professor Nikolai Joukowski (1847–1921) jointly developed the two-dimensional wing theory.

**Two & three-dimensional flow** A two-dimensional airflow covers the items in the following list; all other items are considered three-dimensional flow:

- Lift and drag coefficients, formulas and graphs
- $C_L/C_D$  and graph
- $C_L$  versus angle of attack
- Angle of attack
- Stall and stall angle of attack
- $C_D$  maximum
- $C_L$  maximum
- Induced drag
- Zero lift drag
- Lift and downwash
- Steamlines
- Chordwise pressure distribution
- Centre of pressure
- Stagnation points.

**Two-minute turn** *See* Standard rate turn.

# U

**Ultimate factor of safety** *See Ultimate load factor.*

**Ultimate load factor** All aircraft are designed to withstand a given amount of stress during flight manoeuvres and flight in turbulence up to the design limit load factor. The ultimate load factor allows a 50% safety margin above the limit load factor, to which it is associated to.

It is also known as the ultimate factor of safety.

*See Normal category; Limit & ultimate load factors.*

**Unaccelerated flight** An aircraft in translational, straight and level flight and not turning is in un-accelerated flight.

**Unattached shock wave** This is also known as a bow wave, formed at Mach numbers above Mach 1.2 due to compression on the aircraft's nose heating the air and forming a shockwave to stand-off, or detach, from the aircraft's nose.

*See Shockwaves.*

**Undercamber** A concave curve on the wing's under surface is known as undercamber.

*See Cusp.*

**Units of measurement** The table below shows the units of measurement commonly found in aerodynamics and physics:

|               |                                              |
|---------------|----------------------------------------------|
| Acceleration  | ft/sec <sup>2</sup> or m/sec <sup>2</sup>    |
| BHP           | Brake horsepower                             |
| Calorie & BTU | Heat measurement unit                        |
| Density       | C.G.S. system, grams per cubic centimeter    |
| Density       | mass per unit volume, kg/m <sup>3</sup>      |
| Energy        | Joule                                        |
| Force         | lbf force, kgf force, or Newton              |
| Length        | metre, m, or feet                            |
| Mass          | Kilogram or slug                             |
| Moment arm    | Foot-pound, inch-pound or Newton-meter (N-m) |
| Power         | 1 Joule per second or 1 Watt                 |
| Pressure      | force per unit area                          |
| Temperature   | Kelvin, K, C, or F                           |
| Time          | second, s                                    |
| Work          | N-m (Joules) or feet-lbs.                    |

**Universal Gas Law** *See General Gas Law.*

**Unstable airfoil** It is a requirement for all aircraft to display a limited centre of pressure movement. If this is not so, then the airfoil is considered to be an unstable airfoil.

**Unstaggered biplane** The two sets of main wings are mounted directly one above the other, without any stagger. They are known as orthogonal biplanes.

**Unsteady aerodynamics** It is an alternate name for supermanoeuvrability. The term applies to some jet fighter aircraft capable of manoeuvres at high angles of attack above the normal basic stall angle of approximately 15 degrees.

See **Supermanoeuvrability**.

**Unsteady lift** Unsteady lift is associated with flutter, where rapid vibrations alter the wing's angle of attack and hence, unsteady lift.

**Upwash** The airflow in a subsonic flow has an upward component, known as the upwash, as it approaches a moving wing and then flows up and over the wing's upper surface. A wing at supersonic speed has no upwash; there is insufficient time for the pressure waves to transmit the impending arrival of the wing to the airflow ahead.

See *Diagram 65, Pressure Differential & Stagnation Points*.

**Utility category** Aircraft in the utility category have greater stress limits than normal category aircraft, allowing the aircraft to safely experience up to positive 4.4g and negative 1.8g limit load factors. Basic aerobatic manoeuvres and intentional spins are allowed.

The Beech 77 Skipper and the Piper PA-38 Tomahawk are two aircraft types certified in the utility category. Some aircraft are allowed to operate in the utility category at less than maximum all up weight; the maximum take-off weight must be reduced to 15% less than that allowed in the normal category.

See **Load factor; Limit & ultimate load factors**.



**U-wave** The U-wave is formed in a similar fashion to the N-wave sonic boom (formed in straight and level flight). However, a supersonic aircraft forms the U-wave during flight manoeuvres. The U-wave is a focused boom with shockwaves generated by the front and rear of the aircraft. The peak overpressure is greater in the U-wave than in the N-wave but covers a smaller ground surface area.

*Compare with N-wave.*

# V

**V<sup>2</sup>** The velocity squared as found in the lift and drag formulas:

$$\text{Lift (or drag)} = C_L \text{ or } (C_D) \frac{1}{2} \rho V^2 S$$

Where  $C_L$  = lift coefficient

$C_D$  = drag coefficient

$\frac{1}{2}$  = a constant

$\rho$  = air density kg/m<sup>3</sup> or lbs/cu.ft.

$V^2$  = aircraft speed in m/sec<sup>2</sup> or FPS<sup>2</sup>

$S$  = wing area in square meters or square feet.

**V<sub>A</sub> manoeuvring speed** The V<sub>A</sub> manoeuvring speed is the highest equivalent air speed at which full up-elevator loads the main wing to its maximum limit load factor. At air speeds below V<sub>A</sub>, the wing will stall before reaching its maximum structural limit. Above V<sub>A</sub>, the structural limit will be exceeded. For normal category aircraft, the limit load factor is 3.8g, which should be considered as the maximum safe limit. Pulling more than 3.8g places the aircraft at risk of structural damage, hence the need for a V<sub>A</sub> manoeuvring air speed.

Elevator authority increases with an increase of air speed and a decrease of aircraft weight. Therefore, it can change the angle of attack at an increased rate imposing more loading on the wings. For this reason, V<sub>A</sub> is given only for the maximum all up weight and it reduces with a reduction in weight. Because it varies, it is not marked on the air speed indicator. For a simple rule of thumb, it is twice the flaps up stall speed (V<sub>S1</sub>) – the bottom end of the green arc on the air speed indicator.

The V<sub>A</sub> manoeuvring speed is calculated from the flaps up stall speed (V<sub>S1</sub>) times the load factor. Assume a V<sub>S1</sub> of 48 Knots and limit load factor of 3.8 it follows:

$$\begin{aligned} V_A &= 48 \text{ times } \sqrt{3.8} \\ &= 48 \text{ times } 1.95 \\ &= 94 \text{ Knots.} \end{aligned}$$

For a light aircraft, the difference in weight does not amount to much, may be only 5 Knots difference in  $V_A$ . However, moving up to heavier aircraft the difference in weight and therefore the manoeuvring speed become more significant resulting in a wider speed range between minimum and maximum  $V_A$  speeds.

See **Manoeuvre envelope**.

**Vanes** Vanes are individual airfoils making up a flap system. For example, triple-slotted flaps consist of three separate vanes. The term can also refer to auxiliary airfoils attached to primary control surfaces.

Aerodynamically shaped vanes, which are non-rotating, are used in conjunction with the rotating blades to guide the airflow through a gas turbine (jet) engine's compressor or turbine. They are also known as stator blades.

**Vapor cone** A vapor cone is an alternate name for the Prandtl-Glauert condensation cloud caused by a sudden drop in air pressure in the airflow through a shockwave.

See **Prandtl-Glauert condensation cloud**.

**Variable camber flap** A Krueger type flap, which increases camber when deployed.

**Variable-geometry wing** The variable-geometry wing can be positioned to the appropriate sweepback angle for a chosen speed range. The variable-geometry wing benefits from the best characteristics of both the straight and sweptback wings.

The sweepback angle can be varied in flight from approximately 10 degrees (a straight wing) to about 70 degrees (for a sweptback wing). The straight wing's position produces a high aspect ratio wing with a high maximum lift/drag ratio ideal for slow flight during take-off and landing. Sweeping the wing back to a medium

sweep angle is the ideal position for high-subsonic cruise and a maximum sweep angle is used for supersonic flight.

Transonic and supersonic flight requires a thin, low aspect ratio wing to reduce drag. Increasing the sweepback angle reduces the wingspan and decreases the thickness/chord ratio, which in turn increases the critical Mach number and gives a lower maximum lift/drag ratio. At speeds higher than Mach 1.0, the overall drag decreases significantly with increased sweepback angle. Another advantage of sweepback on supersonic aircraft is the decreased wingspan, which keeps the wing within the Mach cone generated at the nose of the aircraft or the wing root leading edge.

In order for the wing to be able to swing back and forth, the wing is mounted on pivots. A single-pivot point mounted on the aircraft's centreline is very unsuitable. The reason is, as the



**Photo:** The Bell X-5 research aircraft, seen here at the National Museum of the United States Air Force at Dayton, Ohio, was the first aircraft to flight-test the variable geometry or swing wing concept in flight.

wing swings rearwards the wing's centre of gravity and centre of lift would move rearwards, with the centre of lift moving a greater distance rearward from the centre of gravity. This would increase the aircraft's longitudinal stability, which under normal circumstance would be ideal. However, with fighter aircraft too much stability reduces manoeuvrability – an adverse requirement. Another disadvantage is the increase in trim drag.

To overcome the problems of a centreline single pivot, dual outboard pivots – one for each wing – were developed in 1959 by NASA Langley. They allow the wing roots to translate forward with aft swing. The wing's weight is balanced more evenly with less movement of the centre of lift and centre of gravity. Therefore, longitudinal stability and manoeuvrability remain the same as for an unswept wing.

The large, streamlined, and highly swept cuffs attached to each side of the fuselage, which covers the pivot points, often identify aircraft with variable-geometry wings. The cuffs were developed at the NASA Langley Research Centre.

The idea of variable-geometry wings has been around since World War II. The German Messerschmitt P.1101 was designed as a variable-geometry aircraft but it never flew. After the war in 1945, Barnes Wallis (1887–1979) of the Vickers Aircraft Company in the UK brought the idea a stage further. With the help of a NASA wind tunnel, the Bell X-5 research aircraft was the first aircraft to flight-test the variable-geometry concept, with Bell's chief test pilot Jean 'Skip' Ziegler (1920–1953) at the controls, with a first flight on 20 July 1951. The Bell X-5's sweep angle range varied from 20 to 60 degrees sweepback.

The Grumman XF10F designed as a variable-geometry fighter, followed the Bell X-5; although the variable-geometry wing worked well, it never entered military service. However, the first operational variable-geometry fighter to enter operational service was the now well-known General Dynamics F-111 fighter-bomber developed jointly by the Vickers Company and NASA. The wings

sweep through a range from  $16^{\circ}$  to  $80.5^{\circ}$ . The F-111 entered service on 21 December 1964 with the USAF. The Russian Tupolev TU-160 jet bomber is the world's largest variable geometry aircraft. At an all up weight of 280 000 kilograms (605 000 pounds), it is larger and heavier than America's Rockwell B-1 Lancer swing-wing bomber.

Other variable-geometry fighter aircraft have entered service since then. However, due to the maintenance complications and extra weight of the pivot system, the swing-wing concept has been phased out since the 1980s.

A common name for a variable-geometry wing is swing-wing.

**Variable incidence tailplane** The variable incidence tailplane is now found on all modern jet transport aircraft, fighters, and a few light aircraft, such as the Piper PA-28 Cherokee series. Trim drag is reduced with the tailplane at the most suitable angle of incidence to keep the elevator streamlined with the airflow.

The single-piece main control surface is used to trim the aircraft and is used for pitch control.

See **Stabilator; All-flying tailplane; Slab tailplane.**

**Variable main wing incidence** As the name suggests, variable main wing incidence allows the main wing to be hinged upwards to increase the wing incidence relative to the fuselage. This method is used to advantage on the Chance Vought F-8 Crusader for example, where the increased angle of incidence provides STOL performance for take-off and landing. A second advantage on approach to land is the fuselage pitch attitude is lowered improving the view ahead for the pilot.



**Photo:** The Chance Vought F-8 Crusader has a variable incidence main wing. This Crusader is located at the Pima Air & Space Museum, Tucson, Arizona.

**Variable-pitch propellers** A variable pitch propeller consists of two or three adjustable variable-pitch positions. The propellers can be classified under two basic headings of ground adjustable type, and in-flight controllable props. They have two (or three) variable pitch positions of fine pitch and course pitch, (or also feathered). A variable pitch propeller does not have a constant speed unit and therefore it maintains a constant RPM in the same manner as a fixed-pitch propeller varying RPM with air speed and power changes.

**Vectored thrust** All alternate method to V/STOL operation is achieved by using vectored thrust, where strategically placed nozzles direct jet efflux in the required direction to manoeuvre the aircraft in the hover and transition to and from forward flight.

The most famous V/STOL aircraft to be built and see military service is the Hawker/BAC/McDonnell Douglas Harrier 'Jump Jet', which achieves its versatility due to its Rolls-Royce Mk 105 Pegasus vectored thrust turbofan engine. The Harrier was developed from the experimental Hawker Siddeley P.1127 Kestrel with its Bristol BE.53 vectored thrust engine, built in 1957.



**Photo:** The BAC Sea Harrier is famous for its vectored thrust (VTOL) technology and is shown here at the Newark Air Museum at Newark-on-Trent, England.



**Vectors** A vector is an arrow drawn on a diagram showing direction and magnitude (size) combined for a given force. Vectors are used in many aerodynamic diagrams. Vectors can be added or subtracted. A coordinate plane shows the vectors as positive when drawn upwards, and/or to the right, and negative when drawn downwards, and/or to the left.

*See Parallelogram of forces.*

**Velocity** Velocity, either in a straight line or angular as in a turn, is a measure of the speed of a vehicle (aircraft) plus its direction of travel. Any change in velocity is referred to as an acceleration if positive, or negative acceleration if decreasing. The term 'velocity squared' ( $V^2$ ) appears in the lift and drag formulas.

**Velocity/drag ratio** The ratio of the aircraft's velocity to its drag is known as the velocity/drag ratio.

**Velocity of advance** The velocity of advance is equal to the propeller propwash velocity but ignoring the propeller tip's RPM. It is greater than the aircraft's forward speed while under power from the engine.

**Velocity slip** In a subsonic boundary layer flow, the air particles in contact with the body surface become stationary. However, in a low-density flow as found at very high altitude, friction drag becomes negligible and so the boundary layer flow does not become stationary anywhere in the layer. This is known as velocity slip and becomes of significance in hypersonic flow.

**Vena contracta** The vena contracta is the position in the propeller propwash – after it has passed through the propeller disc – where the airflow's stream tube contraction and velocity are greatest. It occurs downstream at a distance equal to the propeller's diameter. In accordance with the conservation of mass, which

remains constant in a subsonic flow, the airflow velocity increase through the propeller disc causes the stream tube to contract.

The term *vena contracta* is Latin for 'contracted vein'; Evangelista Torricelli (1608–1647) introduced the term in 1643.

## **Ventral fairing** *See Belly fairing.*

**Ventral fin** A single ventral fin is located on the underside of the aircraft below the vertical tail. It is the equivalent to the dorsal fin located on top of the fuselage. A ventral fin is associated with the aircraft's longitudinal stability its purpose being twofold:

- first, it increases the area of the vertical fin providing greater stability in the yawing plane. On a swing-wing aircraft with the wing swept aft for high-speed flight, the centre of gravity moves aft reducing the couple between the centre of gravity and the fin, reducing yaw stability. This is where the ventral fin comes into play by providing greater fin area to restore the reduced stability and thus increase its yaw damping ability.
- second, the vertical fin placed above the fuselage in the conventional manner has an effect on the placement of the OX longitudinal axis. It is commonly assumed the OX longitudinal lies between the aircraft's nose and tailcone, but this is not strictly true. The fin has the effect of raising the axis at the rear of the fuselage so that it slopes upwards from nose to tail; the upslope is de-stabilizing. This is where the ventral fin provides its secondary effect; it tends to lower the axis at the rear placing it in a more horizontal attitude and thereby improving yaw stability.

It follows, if the vertical fin was placed below the fuselage instead of on the top, the stability would be increased. However, the fin would be a hindrance during rotation on take-off and landing. The Russians solved the problem of the ventral fin's position on the Mikoyan Gurevich MiG-27-D Flogger swing-wing fighter-bomber by hinging it to lie horizontal during take-off and landing; it is then lowered to hang vertically during cruise flight. McDonnell used a

different approach to solve the same problem on their F4-Phantom II supersonic combat aircraft by using an anhedral tailplane.

A ventral fin may also be found on some light aircraft; its purpose there is usually associated with improving spin characteristics.

**Ventral strakes** Two plates (strakes) attached to the rear, underneath of the fuselage in the ventral position and angled slightly outboard to provide yaw damping and enhance directional stability.

**Venturi** The venturi is a tubular shaped device with a convergent/divergent throat for changing the pressure in a fluid flow (air).

Commonly, the venturi is operated in subsonic, incompressible airflow where the airflow, on approaching the restriction increases



**Photo:** A venturi tube mounted on the side strut of a DH 89 Dominie biplane of 1934 vintage.

its speed with a resulting decrease in pressure. Once past the constriction, the airflow velocity decreases accompanied by an increase in pressure. This is used to advantage for example in piston-engine carburetors and suction powered flight instruments on early light aircraft.

A supersonic, compressible flow through a venturi type device – known as a de Laval nozzle – acts in the reverse mode to a subsonic, incompressible flow, where the velocity initially decreases slightly due to the shockwave and after passing through the restriction into the diverging section, the airflow velocity increases further and the pressure decreases.

The 19<sup>th</sup> century Italian physicist, Giovanni Battista Venturi (1746–1822) introduced the venturi. It was used on an aircraft for air speed measurement by the Frenchman, Captain A. Eteve in 1911.

*See de Laval nozzle; Convergent/divergent duct; Compression flow; Expansion flow & fan.*

**Vertical axis** The OZ normal or vertical axis is perpendicular to the OY lateral and the OX longitudinal axes and passes through the centre of gravity.

*See Axes.*

**Vertical stabilizer/tail** The Americans prefer the term vertical stabilizer; otherwise, it is known as the vertical tail, fin, or vertical fin. Its area must be greater than the fuselage side area forward of the centre of gravity to provide directional stability. In addition, a multi-engine aircraft requires extra fin area to counter asymmetric yaw due to engine failure.

*See Tailplane.*

**Viscosity** The viscosity of a fluid (air) refers to the ratio of shear stress to the strain in the fluid, or more simply the ratio of inertial forces to viscous forces. It is the ease, or difficulty, with which a fluid will flow due to internal friction. Inertial forces are those, which are resistant to change, and viscous forces can be described as sticky.

The viscosity of air can be considered to increase with a temperature rise but is virtually unaffected by the air pressure, as opposed to a liquid fluid where the temperature and pressure must be considered. Its importance is relevant in air friction (drag) and aerodynamic scale effect calculations. The Reynolds number (Re) is the dimensionless ratio of inertial force to the kinematic viscosity force.

The coefficient of absolute viscosity is denoted by the Greek symbol of  $\mu$  (mu). However, in aerodynamics a more useful term is the kinematic viscosity, denoted by the Greek symbol of  $\nu$  (nu), the formula being as follows:

$$\text{Kinematic viscosity} = \frac{\text{coefficient of absolute viscosity}}{\text{density}}$$
$$= \nu = \mu/\rho.$$

Kinematic viscosity of air at sea level in ISA conditions is equal to 0.000158 square feet per second and increasing to 0.000349 at 30000 feet altitude. The SI unit of kinematic viscosity is  $\text{m}^2/\text{s}$ .

Shear viscosity is an alternate name.

*See Reynolds number.*

**Viscous dissipation** Aircraft designed to fly at hypersonic speeds have blunt nose bodies to dissipate some of the airflow's high temperature incurred during hypersonic flight; this is known as viscous dissipation.

*See Hypersonic aerodynamics.*

**Viscous drag** *See Skin friction drag.*

**Viscous flow** Both laminar and turbulent boundary layers consist of a viscous flow as opposed to an inviscid flow outside of the boundary layer. Viscous flow can be considered as 'sticky' or a friction drag producer.

In aerodynamics, we can assume an inviscid flow exists for convenience when studying air pressure and lift distribution about an aircraft in flight, although a true inviscid flow does not actually exist in reality. However, viscosity must be taken into consideration when studying scale effect, airflow and the effects of friction drag. At high Mach numbers, the kinetic energy in the viscous flow transforms into internal energy, which is the cause of the temperature rise and a density decrease within the boundary layer.

*Compare with Inviscid flow. See Diagram 89, Transition Point & Region.*

**V-n manoeuvre/gust envelope** *See Manoeuvre envelope.*

**Von Karman ogive** *See Sears-Haack body.*

**Von Karman street** *See Vortex street theory.*

**Volkenrode Research Institute** Volkenrode was an aeronautical research institute in Braunschweig, Germany, during WWII. It was the largest and most modern facility in Germany by the end of the war and unknown to the allies until discovered by advancing US ground forces in May 1945. Numerous secret papers on aeronautical research were discovered there. In particular, papers on swept wing aircraft designed to fly through the transonic region were discovered, which helped the USA in the development of jet aircraft. The Boeing Company redesigned the engine pods for its new Boeing XB-47 Stratojet jet bomber in 1947, due to the information gained from the papers discovered at Volkenrode.

**Vortex breakdown or burst** Sweptback, low aspect ratio wings increase their maximum lift coefficient at high angles of attack by the formation of vortices over the wing's upper surface. Lift is increased substantially with the presence of the vortices. However, just like a laminar flow over the wing that breaks down at high angles of attack, so can a vortex. A maximum angle of attack can be reached where the steady vortex breaks down resulting in a loss of most, or all of the lift induced by the vortex. It is caused by an adverse pressure gradient destroying the vortex – and so it bursts.

**Vortex filament** A vortex filament is akin to a streamline, but has vorticity.

Hermann von Helmholtz (1821–1894) was instrumental in vortex research.

**Vortex flow & drag** A vortex flow combines translation and rotary motion (vorticity). A very good example of vortex flow is found in wingtip vortices, especially when humid air conditions make them visible. Vortex flow is also used to advantage over delta wing roots to intensify lift at high angles of attack. The vortex flow produces drag but generally the vortex is downstream of the aircraft and its drag is considered to have no effect on the total aircraft drag.

The term vortex is singular; vortices are plural.

**Vortex generator farm** A row of vortex generators spaced a short distance apart in a more or less straight line are used for example on the wing's upper surface to re-energize a degrading boundary layer.



**Photo:** A vortex generator farm on the outer wing panels of a Douglas A4-K Skyhawk attack aircraft.

**Vortex generators** Vortex generators are low-aspect ratio airfoils mounted usually, near the wing's point of maximum thickness, on the tail fin, or other parts of the fuselage. They are mounted at an angle, of about 15 degrees to the airflow, not streamlined, and project about 2–3 centimeters (1 inch) into the airflow boundary layer. They form mini vortices, co-rotating – all the same way, or contra rotating – in opposite directions. Vortex generators re-energize sluggish boundary layer airflow and increase its velocity by swirling and mixing the slower moving air within the boundary layer with the higher speed of the laminar



flow above the turbulent boundary layer. The separation of the boundary layer particularly at high angles of attack is delayed. They also help to dissipate or reduce upper surface shockwaves and reduce the strength of the wave drag. Although they create a little drag themselves and do nothing to improve lift, they reduce the effects of the adverse boundary layer drag and separation, which is far greater. They can also reduce airframe buffeting and improve the airflow over control surfaces and flaps. The Boeing 737 jet transport has a few vortex generators mounted on the nose below the windshield, which reduces cockpit noise levels by 2–3 decibels and two rows of generators (vortex generator farms) on each main wing for the reasons described above.

Vortex generators were flight-tested on the Douglas D-558-1 Skystreak research aircraft.

They are also called turbulators.

See **Boundary layer control system**.

**Vortex hazard** Wingtip vortices trailing from the main wings produce a very strong vortex, which can be a hazard to following lighter aircraft, especially during approach to land and just after take-off.

It is also known as wake hazard or wake turbulence.

**Vortex lift** Vortex lift is a phenomenon peculiar to delta wing aircraft with a sharply defined, highly swept wing leading edge.

Wingtip vortices are present on all wings during their lifting process. The tip vortices on delta wings intensify with increasing angle of attack and move inboard along the leading edge towards the wing roots enveloping the whole wing in a series of conical vortices, which combine and roll up to form a large vortex over the upper surface of the wing. The leading edge vortices (vortical flow) rotate in a helical flow in the opposite direction to the trailing edge vortices.

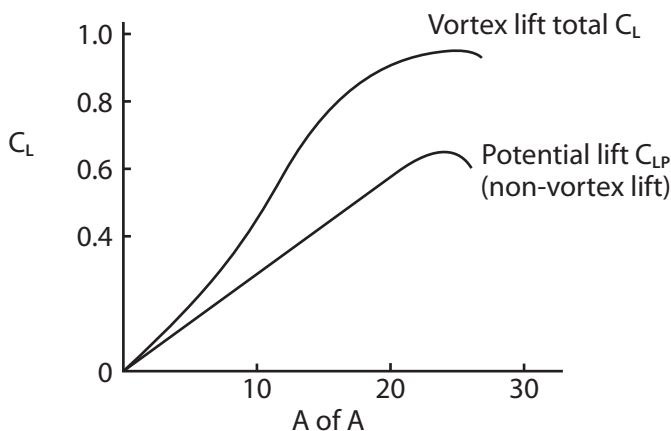


**Photo:** The Handley-Page 115 delta wing research aircraft was built to flight test the vortex lift phenomenon on delta wing aircraft. This HP-115 is displayed at the Fleet Air Arm Museum, Yeovilton, UK.

The increased, high velocity, reduced upper surface low-pressure area created by the vortices increases the lift coefficient in a non-linear manner and hence, increases the lift. The lift/drag ratio is approximately doubled at high angles of attack. The vortex core stays close to the wing's upper surface and generates the extra lift required during low-speed, high angle of attack flight. The

difference between the normal wing lift and that produced by the vortices accounts for the vortex lift. The slenderness of the wing determines the angle of attack at which the vortices commence. This is known as the Polhamus suction analogy.

On delta wings with a sweep back angle of less than about  $55^\circ$  the vortex is weaker and produces some buffeting. With greater sweep back angles, the vortex is more stable with less buffeting due to the greater vorticity of the vortex core. On Concorde's wing, the action of the vortex can be felt as a mild bouncing just after take-off and during the final approach to land when the aircraft is generating vortex lift.



**Diagram 92, Vortex Lift Coefficient V. Angle of Attack**

Vortex lift lowers the minimum flight speed and postpones the stall angle, although to be precise delta wings do not stall in the normal sense, but reach an angle of attack where the rate of climb becomes zero due to the very high drag; this is considered the delta wing's equivalent to the stall.

The vortex lift core rotates much slower than a normal wingtip vortex; therefore, wake turbulence is less of a problem to following smaller aircraft. Dr. Dietrich Küchemann (1911–1976) and E. C. Maskell discovered the vortex lift phenomenon.

Test pilot Chuck Yeager first flew the delta wing Convair XF-92A research aircraft on 18 September 1948. On his second landing, he noticed the phenomena of vortex lift. A few years later, the Handley Page HP 115 research aircraft with slender delta wings and fixed undercarriage proved that vortex lift was viable at low-speeds and would be an asset for Concorde. Its first flight was made in August 1961 and it was based at RAE Bedford for research in low-speed, high angle of attack flight during the development of Concorde's wings.

It is also known as the three-dimensional (3-D) vortex lift theory. See chines, separation, forebody strakes, form thrust, slender wings, and Polhamus suction analogy.

**Vortex line** On days of relatively high humidity, the trailing wingtip vortices become visible due to the moisture condensing in the low-pressure core; the visible trail is known as the vortex line.

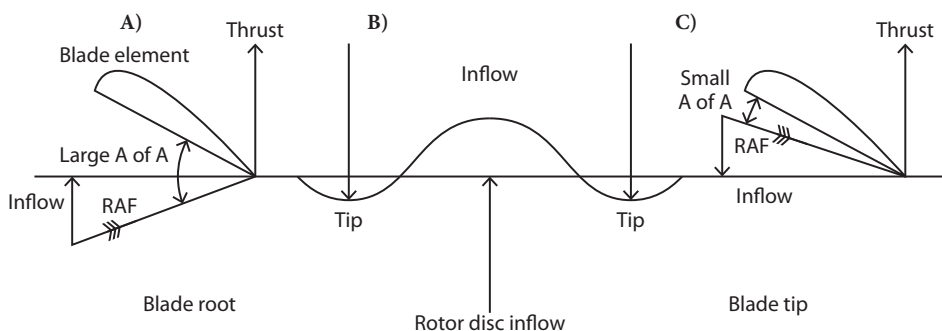


**Photo:** Visible vortex lines also emanate from propeller tips under humid air conditions, or rain as in this photo of a Bombardier Q-300 turboprop.

**Vortex ring state** One mode of flight that all helicopter pilots prefer to avoid is vortex ring state. It involves the downwash through the helicopter's rotor disc when making a descent to land with a rate of descent at about 300 feet per minute, with a low forward speed, and a tail wind or vertical descent.

**Diagram 93, Vortex Ring State**, shows an opposing inflow pattern during the vortex ring state. Initially, as the helicopter commences its descent, the inflow enters the rotor disc area from above, producing an average angle of attack along the blade length, being slightly greater at the blade roots and less at the tips.

With the descent progressing, the angle of attack at the rotor blade roots increases until eventually the blade roots stall, (**Diagram 93A**) and the stalled area gradually spreads along the blade towards the tips. The inflow in the blade root area changes from a down flow to becomes an upward flow, relative to the helicopter's descent and opposite the downward flow at the blade tips (**Diagram 93C**). At the blade tips, the angle of attack reduces as the inflow approaches from above in the normal way. The result is the blade tips are operating at a relatively small angle of attack and not providing very much lift, while the blade roots are stalled and do nothing to aid the lift. **Diagram 93B**, shows the rotor disc with stalled blade roots (at centre) and the rotor blade tips operating at a small angle of attack and producing little lift.



**Diagram 93, Vortex Ring State**

A secondary vortex ring forms at the junction of the up and down inflows through the disc causing severe vibration, pitch, roll, and yaw in a random fashion. Rotor efficiency is reduced in the conflicting airflows and the helicopter continues to descend. Increasing power to arrest the rate of descent has the effect of further increasing the vortex ring state and the rate of descent increases even more. The result is a heavy landing – or worse!

The tail rotor may also suffer the effects of vortex ring state when hovering or hover turning. The tail rotor draws inflow air from the right-hand side of the rotor and if a wind is blowing towards it from the left side, a vortex ring state may develop due to the conflicting airflows.

In the English winter of 1946–1947, RAE Farnborough investigated and discovered the reason for vortex ring state by using a Sikorsky R-4 helicopter.

It is also known as settling with power, and recirculation.

**Vortex street theory** The complex vortex street theory was first introduced by Professor Theodore von Karman (1881–1963) and his colleague in 1911.

The theory assumes a vortex sheet replaces the wing surface. The circulation is found by calculation and hence lift is found by the Kutta-Joukowski theory. It deals with the alternating vortices departing the wing's trailing edge from the upper and lower surfaces and the airflow around cylinders and bracing struts, etc. The individual vortices break away in an alternating pattern similar to the staggered lights along a city street, hence the name. The vortex street airflow is what accounts for the whistling noise known as 'singing in the wires'.

Also known as the von Karman vortex street, vortex street, vortex shedding or vortex trail.

**Vortical flow** *See Vorticity.*

**Vorticity** Vorticity is the angular velocity of a uniformly rotating airflow within a viscous boundary layer. Inviscid flows outside of the boundary layer are free of vorticity.

Vorticity =  $2\omega$  = 2 times angular velocity.

Also known as rotational velocity or vortical flow.

See **Vortex lift**.

**Vorticity interaction** This phenomenon is associated with hypersonic flows, where strong vorticity with large velocity gradients are induced in the flow field of the normal boundary layer, and the entropy layer.

See **Entropy layer**.

**Vortillon** A vortillon is an aerodynamic chord-wise fence type of attachment located on the forward, upper surface of the wing, and wrapping around and under the leading edge. The vortillon reduces the spanwise boundary layer airflow towards the wingtips, a common occurrence on sweptback wings. Wrapping around the wing's leading edge it protrudes into the stagnation line inducing vortices to form and re-energize the airflow over the wing. Thus spanwise flow was reduced enticing the wing root area to stall before the outer wing panels causing the nose to pitch down to recover from the stall. Vortillons also reduced wing downwash over the T-tail avoiding a potential deep stall.

The Douglas Corporation introduced the vortillon when they designed and built the Douglas DC-9 jet transport, which first flew on 25 February 1965.

The vortillon is also known as a wing fence.



**Photo:** A vortillon mounted on the wing leading edge of a Douglas DC-9 airliner. This DC-9, located in the National Air & Space Museum, Santiago, Chile, is awaiting restoration; this explains the vortillon's close proximity to the ground, considerably lower than normal.

**V-speeds** Reference speeds, known as V-speeds, used by pilots, and frequently referred to in aerodynamics are listed here:

- $V_1$  = take-off decision speed
- $V_2$  = take-off safety speed
- $V_A$  = design manoeuvring speed
- $V_B$  = turbulence penetration speed
- $V_C$  = design cruise speed (top of green arc) 33 times  $\sqrt{W/S}$
- $V_D$  = design dive speed 1.4 times  $V_C$
- $V_{FE}$  = flaps extension speed
- $V_{IMD}$  = minimum drag speed
- $V_{MCA}$  = the minimum control speed (air)
- $V_{MCG}$  = the minimum control speed (ground)
- $V_{ME}$  = maximum endurance speed
- $V_{MO}$  = Maximum operating speed



- $V_{MR}$  = maximum range speed
- $V_{NE}$  = never exceed speed (red line, top of yellow arc) 90% of  $V_D$
- $V_{NO}$  = Maximum structural cruising speed (greater than or equal to  $V_C$ )
- $V_{S1}$  = flaps up stall speed
- $V_{SO}$  = flaps down stall speed
- $V_X$  = best angle of climb speed
- $V_Y$  = best rate of climb speed, minimum drag speed
- $V_{ZRC}$  = zero rate of climb speed with one engine failed
- $V_{ZRE}$  = zero rate of climb speed for a delta wing aircraft.



**Photo:** This French built Fouga CM.170 Magister (located at the Pima Air & Space Museum, Tucson, Arizona) has a very distinctive V-tail or butterfly tail. Some authorities claim the CM.170 to be the World's first two-seat, turbojet trainer, with a first flight on 23 July 1952.

**V-tail** A tailplane empennage combining the vertical fin and horizontal tailplane as one vee-shaped, large dihedral tail unit is known as a V-tail.

The advantage of a V-tail empennage is its reduced weight, less drag, and it also keeps the tailplane out of the trailing vortex wake from the main wings, which reduces buffeting, and it reflects incoming radar signals to make it less visible to enemy radar. However, the disadvantage is, it places greater stress on the tailplane attachment points, and it can suffer from asymmetric tail flutter and cause the aircraft to experience Dutch roll. In addition, the combined pitch/yaw controls, known as ruddervators, are more complex to build.

A modified Hanriot H-28 trainer flight-tested the first V-tail empennage designed by Polish engineer Georges (Jerzy) Rudlicki in 1931 becoming the first V-tail aircraft to fly. Also known as a butterfly tail, it is found on the very popular Beech V-35 Bonanza light aircraft, the French Fouga CM 170 Magister jet trainer, and the Lockheed F-117A Nighthawk stealth strike aircraft, to name three examples.

See **Butterfly tail**.



**Photo:** The Hawker P.1127 was a forerunner of the BAC Harrier. An example is shown here at the Fleet Air Arm Museum at Yeovilton, England.

**VTOL** VTOL is the acronym for vertical take-off and landing. Several aircraft have been built and tested over the years to study the problems of vertical take-off and landing. One of the earliest aircraft was the McDonnell XV-1, which made its first vertical flight in February 1954. Since then, over forty aircraft have been built in an attempt to develop a VTOL aircraft.

Dr Alan Arnold Griffith (1893–1963) was instrumental in early jet power research and he became the chief scientist for Rolls-Royce jet propulsion. He is noted for the development of the Rolls-Royce Avon and Conway jet engines and the vertical take-off Rolls-Royce Thrust Measuring Rig (Flying Bedstead), in 1954. The research gained was later used on the Short SC.1, Britain's first experimental fixed-wing VTOL aircraft, a forerunner to the British Aerospace Harrier jet.

# W

**Wake hazard** The wake is the disturbed region of turbulent air flowing off the wing's trailing edge.

The wing's vortex sheet, wing tip vortices, jet engine exhaust, propwash, and the disturbed flow over the entire aircraft, all contribute to the wake. However, it is the wing tip vortices causing wake turbulence, which have the greatest effect, and hazard to following aircraft.

**Wake turbulence** Wake turbulence is caused by the rotating vortices (like a horizontal tornado) shed from the wingtips of an aircraft in flight.

The vortices are a product of wing lift and commence when an aircraft rotates on take-off and the wings start to produce lift; the vortices cease on touchdown. Due to the airflow over and under the wing meeting at the trailing edge at an angle (known as the rake) it causes trailing edge vortices, which combined with the wingtip vortices, can be very powerful on large aircraft. The heavier the aircraft, the stronger are the vortices, which should be avoided at all times by following light aircraft, especially on take-off and landing.

Close to the ground, the vortices are weaker than in flight at altitude due to their interaction with the ground, but are still strong enough to upset aircraft of lighter weight than the aircraft producing them. To avoid wake turbulence on take-off behind another aircraft, the pilot should lift-off at a point prior to the other aircraft's lift-off point and stay above and upwind of its vortices. On landing, touchdown beyond the other aircraft's touchdown point. The landing separation distance is increased for aircraft following 'heavy' aircraft (with all up weights in excess of 300 000 pounds). The Boeing 757 is noted for the greater wake turbulence it produces, compared to other heavy aircraft of greater weight.

Also known as vortex hazard or wake hazard.

See **Groundwash; Induced drag.**

**Wallis, B.** Barnes Neville Wallis (1887–1979) was an English engineer and inventor. He was credited with designing the R100 airship, the bouncing bomb of Dambuster fame, and the geodetic fuselage construction used on the Vickers Wellesley (which evolved into the famous Wellington bomber of WW II). Also to his credit is the development of the swing-wing or variable geometry wings for supersonic aircraft.

**Wallowing** An aircraft's successive oscillations around all three axes is known as wallowing.

*See Dutch roll.*

**Wash-in** Inbuilt geometric wing-twist resulting in the angle of incidence increasing towards tips, places the wingtips at higher angle of attack than the wing roots.

Wash-in is not very common on modern aircraft. However, it was used on early aircraft to counteract the effects of propeller torque and rotary engines.

**Wash-out** The decrease in angle of incidence of the wing towards the wingtip (or rotor blade tip) is known as wash-out.

The reason for wash-out is to reduce the wing's structural weight and the angle of incidence at the tip compared to the root incidence in order to control the stall and induce the wing root to stall before the tip. With the root stalling first, the stall break should develop straight ahead with the nose pitching downwards on aircraft with straight wings and without any tendency to roll off to either side.

Wash-out is built into the wing intentionally as a part of its structural design, but it can also manifest itself undesirably, especially in sweptback wing aircraft. During a turn, the wing loading is increased and the wingtips are raised up, twisting the wing's outer section leading edge down, causing less lift. The wing's centre of pressure then moves inboard and forwards reducing the lift/weight moment, allowing the plane to pitch nose-up into a stall.

The sweptback wing's lift distribution is affected by the downwash velocity, which in turn affects the wing's lift. The lift coefficient reduces with wash-out. Without wash-out, a swept wing would have greater lift at the tip and less at the wing root compared to a non-swept wing. Installing wash-out tends to reduce the wing tip lift and the associated structural loads, which induce wing tip stalling. However, it is a requirement for the ailerons to retain their effectiveness into the initial stall break, if possible.

See **Aileron reversal**; **Flexural washout**.

**Wasp wasting** See **Area rule**.

**Wave drag** The combined effects of boundary layer separation drag and energy drag in a transonic flow cause wave drag with a shockwave being the main contributor.

Taking the parasite drag curve from *Diagram 20, Total Drag Curves* and enlarging it to become *Diagram 94, Wave Drag*, it shows the parasite drag is increased in the area of boundary layer separation.

The boundary layer separation drag is caused by the shockwave, which moves aft across the wing disturbing the boundary layer; this is known as the shock-boundary layer interaction. With increasing speed and the shockwave attached to the trailing edge, the boundary layer separation settles down with a reduction of boundary layer drag to its original value. This is shown on the diagram as a bulge on the new parasite dragline; it is known as a transonic drag hump.

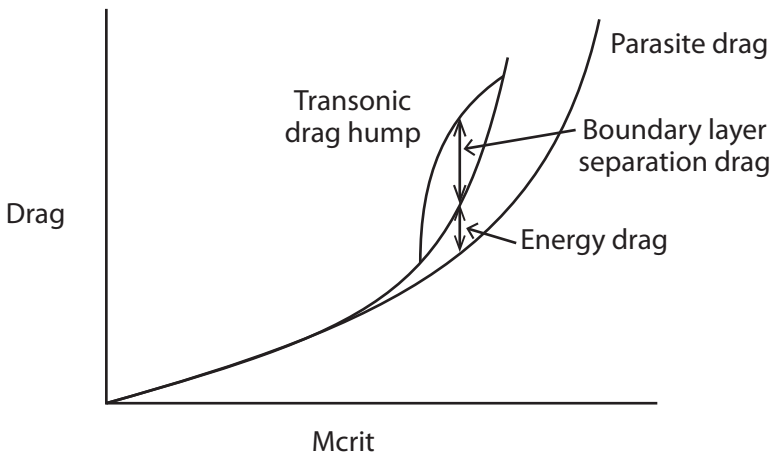
Energy drag is caused when the airflow through the shockwave requires energy to raise the air temperature; the energy is provided by the drag, due to the pressure reduction being greater aft of the shockwave than before it. Due to the temperature rise, normal shockwaves produce greater drag and absorb more energy than do oblique shockwaves, which form in higher flow-speeds. However,

the drag continues to rise due to the oblique shockwave's greater effect in a spanwise direction.

An aircraft on approaching Mach 1.0 has a considerable increase in drag. After passing through Mach 1.2 or slightly higher, the drag reduces to a value close to, but slightly higher than the subsonic value. The wave drag increase is proportional to the wing's thickness ratio squared and inversely proportional to the chord squared. Wings with greater thickness/chord ratios form stronger shockwaves. Wave drag is also approximately proportional to lift squared. That is the reason supersonic aircraft are extremely streamlined with small thickness/chord ratio wings with a sharp leading edge.

Also known as transonic compressibility drag.

See **Drag divergence curve**.



**Diagram 94, Wave Drag**

**Wave riding** Wave riding is another term for compression lift – that extra lift produced by the lower surface of the wing and/or fuselage of an aircraft in high supersonic or hypersonic flight. At very high altitudes used by hypersonic aircraft or re-entry vehicles, the air density is extremely low; however, for wave riding aircraft (or waveriders, as they are known) the low density does become very significant to their method of sustained flight.

*See* **Compression lift.**

**Weak shockwave** *See* **Normal shockwave.**

**Weathercock stability** An aircraft's basic directional stability around the OZ vertical or normal axis is known as weathercock stability. The purpose of the tail fin, which acts like a weathervane, is to keep the aircraft aligned with its direction of flight.

The distance of the tail fin from the centre of gravity and its surface area determines the degree of stability.

Also known as directional static stability.

**Weick, F.** The American, Fred Ernst Weick (1899–1993) invented the NACA engine cowl in 1929 designed for improved engine cooling with less drag. He also wrote a book on aircraft propellers in the 1920s. In 1931, he introduced the '50 foot obstacle clearance limit', which is still in use worldwide as a measure for aircraft take-off and landing performance.

He also designed the famous Ercoupe with interconnected ailerons and rudder and a tricycle undercarriage of which, he was a very keen promoter for safety reasons. Fred Weick eventually joined Piper Aircraft where he designed the successful Piper Cherokee line of light aircraft.

*See* **NACA cowling.**



**Weight** Weight is the force exerted on a given mass by the influence of the earth's gravitational force. Weight is a force acting towards the geometrical centre of the earth.

Weight decreases with distance from the centre of the earth. Because weight is a force due to gravity (symbol 'g'), the units used are the same as for force, i.e., the Newton (N). All objects resting on or near the Earth's surface experience the effects of the gravitational pull of the earth, as weight.

Weight should not be confused with mass; mass remains constant while weight decreases with greater distance from the centre of the earth. However, for most practical purposes in aviation, we can consider weight to remain constant with altitude.

For example, the mass of a Boeing 747 is obviously greater than that of a light aircraft. It is more difficult to move a 747 from rest than it is to move a light aircraft. All aircraft become lighter in weight as they increase altitude (move further from the centre of the Earth) but this factor is only of academic interest and can be ignored by pilots.

Weight is a very important factor in aircraft operations and is therefore described in several ways. For example, for aircraft weight we use empty weight, maximum all up weight, operating weight, maximum take-off weight, landing weight, payload and zero fuel weight, etc.

In many countries around the world, weight is measured in the SI units of kilograms (kg) or grams. Other countries may still use the Imperial system of pounds (lb) and ounces. Newtons is another unit used to measure weight.

The relationship between mass and weight is given by the following:

$$W = mg$$

Where: W = weight  
m = mass kg  
g = 9.81N.

*Compare with Mass.*

**Wenham, F.** The Englishman Frances Herbert Wenham (1824–1908) is noted for his paper on ‘Aerial Locomotion’ delivered to the Aeronautical Society in 1866. Francis Wenham introduced the term of aspect ratio to aerodynamics when he wrote about its effect on the wing planform of finite wings. He proved correctly, that a high aspect ratio planform performs better on relatively low-speed wings. He also explained the forward part of the wing produces the most lift with the lift force located at approximately 25% MAC.

In association with John Moses Browning (1855–1926) of machine gun fame, Wenham was the first to design and build a wind tunnel for research purposes in 1871 in Greenwich, London, England.

**Wetted area/surface** The wetted area refers to all the aircraft’s surface area that would be wet if the aircraft were to be submerged in water. Alternatively, all of the aircraft’s outer surface skin area over which the slipstream flows is the wetted area. Wetted area is part of the drag formula where:

$$\text{Drag} = C_D \frac{1}{2} \rho V^2 S$$

Where  $C_D$  = drag coefficient

$\frac{1}{2}$  = a constant

$\rho$  = air density  $\text{kg/m}^3$  or pounds

$V^2$  = aircraft speed in  $\text{m/sec}^2$  or  $\text{lbs/cu.ft.}$

$S$  = wetted area in square meters or square feet.

Frontal or wing areas are also used for drag calculations in place of wetted area. Each area, wetted, frontal or wing area are all convenient values depending on the required calculation.

**Wetted aspect ratio** The wetted aspect ratio is a useful term for the aircraft designer.

It is taken to be the wing span<sup>2</sup>/wetted area, as opposed to the more common aspect ratio term. The wetted area is that part of the drag formula, which takes into account the parasite or skin friction drag plus the induced drag.

See **Aspect ratio; Wetted area/surface**.

**Wheel fairings** Wheel fairings are used to streamline the airflow over the wheels on fixed-gear light aircraft. They are also known as speed fairings, pants (in the USA), spats (in the UK) or wheel pants, which is now an archaic terms.

See **Fairing**.



**Photo:** The Boeing P-26A Peashooter has streamlined wheel fairings covering the complete undercarriage assembly.

**Whiskers** Weak compression waves forming in a bubble of supersonic airflow are known as whiskers.

*See* **Compression waves**.

**Whitcombe area rule** *See* **Area rule**.

**Whitcomb body** Whitcomb bodies, invented by Richard Travis Whitcomb (1921–2009) a NACA researcher, are streamlined bodies attached to wing trailing edges that act to improve the airflow on high-speed wings. The Whitcomb bodies act in a similar way to area rule.

They are also known as a Küchemann carrots, shock body, or speed bumps.

**Whitcomb, R.** Richard Travis Whitcomb (1921–2009) was an aerodynamic researcher at NACA Langley Research centre from 1943 onwards and he became the most influential post-WWII aerodynamicist.

He is best remembered for his discovery of the area rule in 1951, for which he received the Collier Trophy in 1954 and, in 1956 the NACA Distinguished Service Medal. Whitcomb also introduced the supercritical wing in 1965, which reduces drag in the high subsonic speed region. He also introduced the winglet, which reduces induced drag by up to 14%, and the Whitcomb body. Richard Whitcomb is also credited with many other aeronautical advancements, particularly in high-speed aerodynamics.

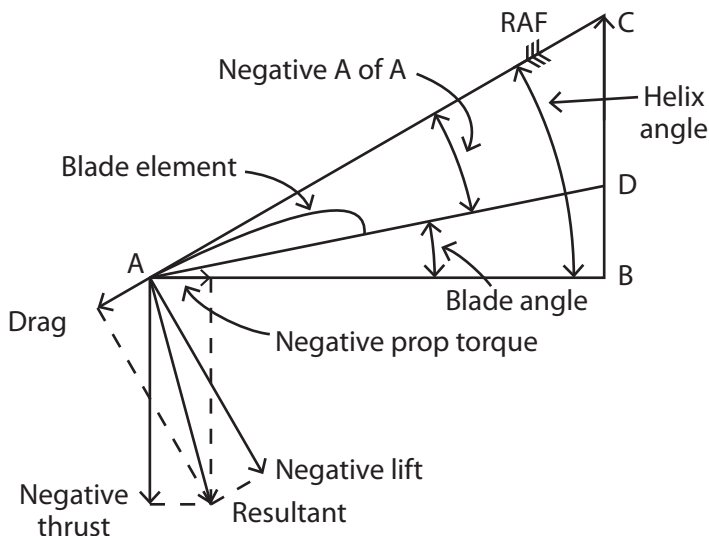
**Wind axis** The aircraft's wind axis is aligned relative to the wind direction drawn through the aircraft's 'O' origin. The flight velocity, angle of sideslip and angle of attack determines the wind axis angle.

**Windmilling propeller forces** In the event of an engine failure in flight, the propeller will continue to turn, or windmill for quite a while after the engine has failed, if it is not feathered.

The force required to turn the propeller after the engine has failed, is provided by the air flowing through the propeller disc due to the forward motion of the aircraft. Referring to **Diagram 95, Windmilling Propeller Forces**, the propeller's rotational velocity (A-B) is assumed to have decreased but the aircraft's forward speed is still relatively high (we assume the aircraft to be in a glide). The vector (A-C) or relative airflow (RAF) will now strike the propeller blade at a negative angle of attack. Assume the propeller pitch to be unchanged for the moment. In this condition, the helix angle will now be greater than the propeller's blade angle and the angle of attack is now negative, the reverse of their normal cruise positions. The negative propeller torque now acts in the direction of propeller rotation powering the propeller to a fast idle speed. With the engine failed, there is no engine torque to oppose propeller torque. If the forward speed is increased even more by lowering the nose, the advance per rev vector (vector B-C) will be extended increasing the negative angle of attack and in turn, increasing the propeller RPM.

By reducing the aircraft's forward speed, the windmilling propeller parasite drag can be reduced. Less drag is of course, an advantage on single-engine aircraft because it reduces the angle of descent allowing more time and a greater distance to be covered in the ensuing forced landing. The reduced windmilling drag on a twin-engine aircraft will lower the  $V_{MC}$  (minimum control speed) enhancing engine-out handling. An inspection of **Diagram 95, Windmilling Propeller Forces**, also shows a reduced forward speed will reduce the advance per rev (vector B-C). The helix angle (AB-AC) will also reduce, thereby reducing the propeller blade's negative angle of attack, which in turn reduces the lift coefficient and the resultant force and hence, reduces the negative thrust (or drag).

The resultant force acting in a rearward direction produces undesirable negative thrust, known as parasite drag or more commonly as windmilling drag. The resultant force is also acting in the direction of propeller rotation. The negative propeller torque component of the resultant force acts in the plane of rotation causing the propeller to continue windmilling.



**Diagram 95, Windmilling Propeller Forces**

Low powered aircraft benefit from a stationary propeller to reduce windmilling drag and improve the glide performance; propeller equivalent flat plate area will be responsible for some drag, however. A stationary propeller is less important on higher-powered aircraft; propeller pitch and throttle setting have a greater effect on glide performance than the flat plate drag of a stationary propeller. A coarse propeller pitch produces less drag than fine pitch and an open throttle is preferred to reduce airflow resistance through the engine; energy is required to overcome airflow resistance, which is obtained from the steeper descent. A stationary or windmilling propeller will always cause the glide angle to be steeper than what the lift/drag ratio would suggest.

**Wind tunnels** Wind tunnels have played an important part in aerodynamic research from as early as 1871.

The wind tunnels are of two basic types – the open ended (with a straight through flow) or the closed circuit type (with a return-flow duct) which are now the preferred type due to their greater efficiency. The working section – where the model or aircraft under test is placed – is classed as an open or a closed section. The open working section allows access to the aircraft/model during tunnel operation as opposed to the closed section type.

Airflow conditions within the modern wind tunnel can be varied to suit the requirements of the test to be performed. Air density, temperature, and velocity can be varied to match the conditions that would be found in the true atmosphere. Increasing the air density in the tunnel reduces the scale effect error between the



**Photo:** This wind tunnel was used to test small models in the early years of flight, circa 1916. It is now on display in the National Museum of the US Air Force, in Dayton, Ohio.

model under test and the full size aircraft. Most tunnels are of the horizontal type; however, some tunnels are mounted vertically for spin testing with an updraught wind flow.

Theodore Theodorsen of NACA was instrumental in improving the open and closed type of wind tunnels.

See **Aeronautical research facilities; Open circuit wind tunnel; Open jet wind tunnel.**

**Wing area** The gross plan wing area denoted by  $S_G$ , includes all the wing area plus that part of the wing upon which the fuselage is located. Unless otherwise stated, wing area is considered the gross wing area. Another term in use is the net wing area, meaning all parts of the wing that is visible; i.e., not covered by the fuselage, denoted by  $S_N$ .

NASA claims the wing area is not the same as total surface area; the total surface area includes both the upper and lower wing surface area. Therefore, the generally accepted meaning of wing area is the plan view area of the main wings, (excluding the tailplane) and is almost half of the total surface area.

‘S’ denotes wing area in the lift (and drag) formula where:

$$\text{Lift (or drag)} = C_L \text{ or } (C_D) \frac{1}{2} \rho V^2 S$$

An alternate name is reference area, denoted by ( $S_{REF}$ ).

See **Gross wing area.**

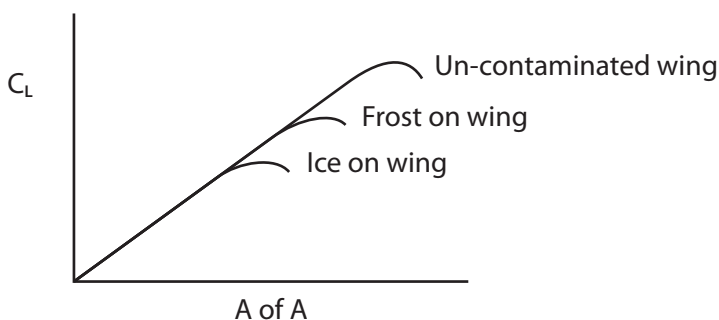
**Wing contamination** The accumulation of ice, frost, snow, or dirt has a detrimental effect on the laminar flow of all aerodynamic surfaces, which includes the wings, propeller, or helicopter rotor blades.

The two major factors of concern are the additional weight and the shape of the ice build-up. The weight of ice on a plane’s wing increases the aircraft’s all up weight, but is of less concern than the increased critical angle and speed. Ice on the leading edge of



helicopter rotor blades can move the centre of gravity forward, which can increase and aggravate the blade's inherent twisting and bending moments.

Moisture in the air and the ambient air temperature determines the amount and type of icing formed. Temperatures just below freezing cause glaze ice, which is heavier than rime ice due to it flowing more freely and smoothly back from the leading edge. Rime ice is formed at lower temperatures and freezes more readily closer to the leading edge. As the ice freezes, air bubbles are trapped within the ice producing a rougher surface compared to glaze ice.



**Diagram 96, Wing Contamination**

The effect of wing leading edge ice accretion has little effect on the lift coefficient at relatively high speeds. However, at lower speeds used for the approach, the ice accretion reduces the lift coefficient, due to boundary layer de-generation, inducing the centre of pressure to move forward and to pitch the nose upwards into a possible stall. The drag coefficient increases and requires greater power to equal the increased drag of the ice accretion. The lift/drag ratio is reduced and hence, performance in general deteriorates.

Leading edge anti-ice or de-ice equipment is applied to the inner portion of the propeller blades and the leading edge of the wings and tailplane. The outer portion of the rotor blades are not so much affected by ice accretion due to the kinetic heating caused by their high rotational velocity at temperatures just below the freezing level. With a decrease in ambient air temperature (to about  $-10^{\circ}\text{C}$ ), the icing spreads along the complete rotor blade length. In normal helicopter flight, the outer portion of rotor blades are the main lift producing area, but in autorotational flight, the inner, driving portion of the rotor blades suffer the most from ice accretion and produce less rotor lift. Therefore, the rate of autorotation becomes greater than normal when ice accretion is present. Ice shedding unevenly from the props or rotor can cause severe vibration through the airframe. The effects of frost, snow, dirt, or sand are of just as much concern as the effects of ice; all contamination has an adverse affect reducing airfoil performance by as much as 30–40%.

**Diagram 96, Wing Contamination**, shows the decrease in lift coefficient due to frost or ice contamination.

**Wing divergence – aeroelastic** Wing divergence is an unwanted, aeroelastic instability problem involving the torsional stress-induced bending of the wingtip area due to aerodynamic forces beyond design limits.

It may occur at relatively high air speeds near the redline speed when the dynamic pressure has increased to reach its maximum value. A point is then reached where the dynamic pressure overpowers the structures torsional strength. Without any warning from buffeting or vibration, the wing is stressed beyond limits in a rearward direction and is torn from the aircraft. In addition, upward stress forces are a problem in the form of an excessive increase in lift, which increases with speed squared acting at the aerodynamic centre, due for example in turbulence, which can increase the angle of attack and lift. The result is, the wing twists further and further until a catastrophic failure occurs when stressed beyond design limits.

Straight wings, low taper ratio, and long span are more prone to wing divergence than sweptback, high taper ratio and short span wings, although aileron reversal is more of a problem on swept wing aircraft due to the thinner wing structure deforming. In high 'g' manoeuvres, the sweptback wing's leading edge may flex downwards reducing lift especially in the wingtip area. The centre of pressure will then move forward causing the aircraft to pitch nose-up.

The extension of flaps can also introduce an aerodynamic twisting moment by increasing the wing's camber. The centre of pressure moves rearwards causing the nose to pitch down creating a twisting force. Although due to the design tolerance this force is negligible, it is still there to some degree.

The aircraft speed at which structural failure occurs is known as the divergence speed.

*See* **Aileron reversal; Flexural aileron flutter; Torsional aileron flutter; Forward sweep wings; Flutter; Aeroelasticity; Divergence; Divergence speed.**

**Wing-drop stall** During a fully developed stall, one wing may drop, which can lead to an incipient spin if recovery action is not initiated by the pilot.

**Wing efficiency factor** The induced drag reduces with increasing speed and increased aspect ratio. When comparing the induced drag produced by straight wings and elliptical wings, it is measured by the wing efficiency factor. Straight wings are less efficient than elliptical wing, producing from 5% to 15% more induced drag. The Supermarine Spitfire WWII fighter has a perfect elliptical wing planform and therefore has less induced drag.

**Wing fence** *See* **Vortillon.**

**Wing gloves** Variable-geometry wing fighter aircraft have wing root extensions on the upper fuselage that covers the swing-wing mechanism – an extension of the pancake; this is known as wing gloves.

**Wing incidence** *See Angle of incidence; Incidence.*

**Winglets** Winglets are located at the tips of the main wings mounted as nearly vertical airfoils, or fins, either canted outwards or toed inwards. They are of two basic types – large or small and both types have their own advantages being unique to each aircraft type and mission requirements. They are a type of non-planar wingtip.

The winglets increase the effective wingspan producing a higher aspect ratio, which results in a higher lift/drag ratio for better aerodynamic efficiency and approximately 4% greater range due to increased fuel economy. With the wingtip vortex moved further outboard, more wing area is available to do the intended job of producing lift. Due to the greater trailing edge length, induced drag is decreased due to the greater spread of wing vortices across the span, which creates extra lift at the wingtip.

By modifying the airflow around the wingtips, the vortex strength is reduced by spreading the wingtip vortex core and reducing its kinetic energy, which improves the airflow circulation and hence, the lift on the outer wing panels. The vortex flow strikes the winglets cambered surface (the inboard surface) which generates a horizontal lift force angled towards the fuselage and slightly forwards. The forward component acts as a forward thrust force opposing the rearward-induced drag force, which is reduced by up to 14%, enabling the aircraft to cruise at higher speeds for the same power or at the same speed with reduced power and a lower fuel consumption and thus extended range. An added bonus is the reduced strength of the trailing vortex reducing the wake hazard to following aircraft.

For short routes, large winglets have the advantage of improving climb efficiency allowing the aircraft to reach the fuel-efficient flight levels in less time. They also improve take-off performance. Small, or raked winglets are more efficient at high altitude (in reducing induced drag) and are therefore more suited to aircraft flying longer routes and spending relatively more time in the flight levels.

Some winglets extent upwards from the rear half of the wing tip only. This is due to the wing's leading edge operating at high Mach speeds, and a winglet located there would experience shock waves and increasing drag, hence the need to mount the winglet away from the tip leading edge.

Winglets have the same advantage as an extended wingspan, but produce less wing root bending moments. Winglet angle varies from the vertical to being tilted outwards. The canted type produce more vertical lift but also place bending loads on the wing, which requires greater strength and hence, greater weight in the structure. The choice of winglet is a compromise between gaining



**Photo:** A Boeing 737 with winglets at Washington Dulles Airport, USA.

the most lift and efficiency for the least wing structural bending loads and weight. Composite materials are used to advantage on modern aircraft with wingtip devices to reduce structural weight and increase strength.

NASAs Richard Travis Whitcomb (1921–2009) introduced the modern version of winglets. Dr. Richard Vogt (1894–1979) also investigated the use of winglets while working for the Boeing Company, which are now common on many Boeing jet transports. However, not all aircraft have winglets because they are not always advantageous. On later models of the Boeing 737 jet transport, winglets are an option to new airline-customers to suit their operational route lengths. The first two large transport aircraft to use winglets from 1988 onwards were the Boeing 747-400 and the EADS Airbus A340.

*See* **End-plates; Blended wingtips; Raked wingtips; Actuating wingtip; Wingtip devices; Wingtip fences.**

**Wing loading** Wing loading is of great importance in wing design and it is a measure of the aircraft's efficiency. The plane's stalling speed is closely related to the wing loading where a low wing loading (measured in lbs/sq.ft) will produce a low stalling speed and vice versa. The aircraft's wing loading is found by dividing the aircraft weight by its wing area.

$$\text{Wing loading} = \frac{\text{weight}}{\text{wing area}}$$

A light aircraft has a wing loading of approximately 10 to 20 pounds/square feet, increasing to higher values of around 70 to 80 pounds/square feet for larger jet transport aircraft.

In straight and level flight, the lift supports the aircraft's weight. During a turn, the lift must be progressively increased to support the weight of the aircraft. Banking the aircraft to make a turn, tips the lift vector towards the centre of the turn decreasing the vertical component of lift. The angle of attack must be progressively

increased to produce the required extra vertical lift. In a sixty-degree angle of bank turn the wings produce twice the amount of lift as required for straight and level flight. Increased wing loading increases the stall speed. In addition, there is a chance for flutter to occur if the wing is not built with sufficient stiffness.

*See Load factor; Turning.*

**Wing load-grading curve** The wing load-grading curve shows the lift pattern over the wing from wingtip to wingtip as viewed from the front of the aircraft.

The most desirable pattern is a semi-elliptical curve but this is not always achieved. Lowering flaps will lift the grading curve higher over the flapped portion of the wing due to the centre of lift moving inboard and increasing. Likewise, lowering an aileron will also raise the lift pattern to show an increase in lift on the outer portion of that wing. Conversely, the opposite rising aileron will produce a depression in the curve showing a decrease in lift over that area.

The wing load-grading curve is also known as the aerodynamic curve, or an elliptical loading curve, due to its shape.

*See Elliptical lift distribution.*

**Wing panels** The wing panels are the outer portion of the wing that attach to the wing's centre section.

**Wing root compression** The wing's thickness/chord ratio varies from root to tip, with the greater thickness at the wing root. A greater thickness/chord ratio provides a greater increase in the airflow's velocity, which can cause wing root compression over the wing root area on low transonic speeds. Weak shockwaves may form and merge into one normal shockwave over the rear of the wing's surface. Area rule is used on some high-speed fighter aircraft to reduce interference drag over the wing root area. No doubt, area rule would also reduce the problem of wing root compression.

*See Thickness/chord ratio.*

**Wing section** The term is an alternate name for the chordwise airfoil section. It is incorrect to apply the term to the wing's centre section (the section of wing attached to the fuselage) or to the outer wing panels.

NASA aerodynamicist, Theodore Theodorsen (1897–1978) introduced the term wing section.

*See* **Airfoil section.**

**Wingspan** The wingspan is the overall distance from wingtip to wingtip, including the fuselage.

The term span is also valid.

*See* **Span.**

**Wing's third region** *See* **Dead air region.**

**Wingtip devices** Wingtip devices can be classified under the headings of wingtips – blended or raked, actuating wingtips, and wingtip fences. The different wingtip devices have different advantages and disadvantages and are designed to suit the aircraft type and its intended mission profile.

They are all intended to modify the wingtip vortices in some way to reduce induced drag, save on fuel consumption, increase range, and improve lateral stability and handling. Some wingtip devices have the effect of increasing the wing's aspect ratio and lift/drag ratio, without physically increasing the span. Other wingtip devices may not have any effect on the lift/drag ratio or any drag reducing properties. Instead, they may create an effective increase in the wing's dihedral, also improving lateral stability. Careful design is required to reduce the inherent induced drag without increasing the wingtip devices profile drag.

From circa 1930s wing root fillets were installed on most aircraft to reduce the interference drag between the wing root and the fuselage. Not much attention was paid to the wingtip design,



which were generally round in planform until circa 1950s, when new aircraft types were introduced with more squared-off tips.

However, as early as 1897 Dr Frederick Lanchester (1868–1946) patented wingtip end plates (a form of wingtip device) to control wingtip vortices. One of the earliest aircraft to sport a wingtip device was the Curtiss SO3C Seamew, a naval reconnaissance aircraft. It first flew in 1940 with an early form of winglet of relatively wide-chord, low height and not sufficiently effective. Having said this, Richard T. Whitcomb (1921–2009) at NASA Langley Research Centre, is credited with ‘inventing’ the winglet during the oil crises of 1973 to reduce drag and hence, fuel consumption. The Gates Learjet Model 28/29 Longhorn business jets were the first modern production jet aircraft to utilize winglets, with a first flight on 24 August 1977. The Learjet’s range was increased by 6.5% and stability was improved.

NASA and the US Air Force used a Boeing KC-135 Stratotanker at NASA Dryden Flight Research Centre to flight test winglets. A Lockheed L-1011 and McDonnell Douglas DC-10 jet transport aircraft were also used for flight testing winglets during 1979–80. The up-dated version of the DC-10, known as the McDonnell Douglas MD-11 was so equipped. Another aircraft of note, to use winglets was NASA’s Boeing 747 Shuttle carrier. On this aircraft, they were mounted on the horizontal tailplane to increase directional stability when carrying the Space Shuttle piggyback.

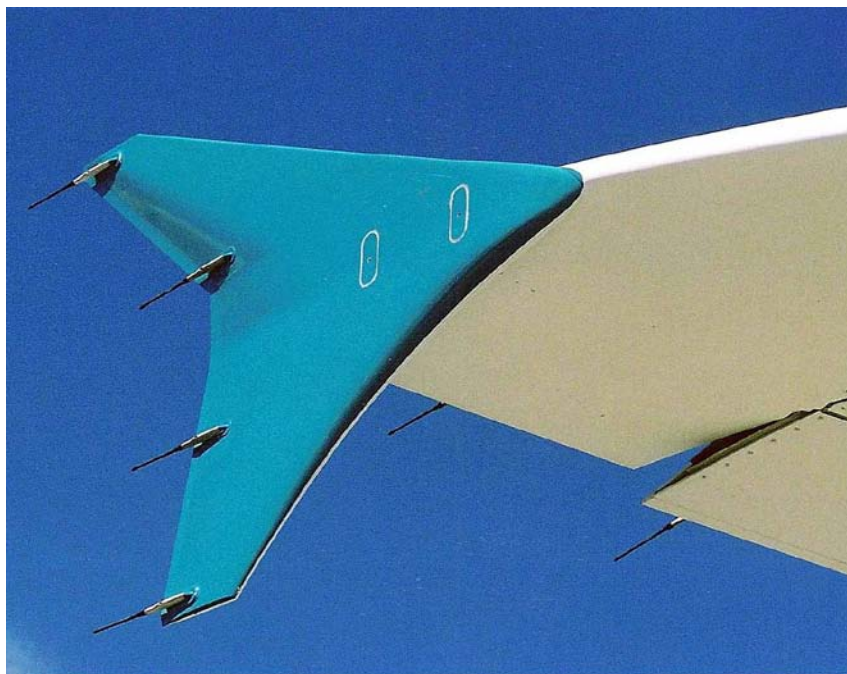
The aircraft designer, Burt Rutan was the first to use winglets on his single-engine VariEze a canard light aircraft. The winglets doubled as tailfins and rudders on the tips of the rear-mounted main wings. The VariEze made its first flight on 21 May 1980. In 1986, the futuristic Beech Starship twin-turboprop, designed by Burt Rutan (1943–20?) was similarly equipped with toed in winglets.

Jeff Jupp and Peter Reeselet, two BAe system engineers, patented wingtip fences. The French EADS Airbus company favors them for their Airbus A310, A320, A318, A350, and A380 jet transport aircraft.

*See* **Winglets; Raked wingtips; Actuating wingtip; Wingtip fences.**

**Wingtip fences** Wingtip fences are a type of delta winglet mounted on the end of the main wings and protrude above and below the wingtip. They are of a similar height to a winglet, or slightly smaller and produce similar benefits in induced drag reduction. Two BAe Hatfield system engineers, Jeffrey Jupp and Peter Reeselet patented them. The first aircraft to employ the wing fences was the Airbus A300-600 jet airliner in 1986; they are becoming common on most jet transport aircraft.

It is also known as an endplate winglet.



**Photo:** A wingtip fence is shown here mounted on the wingtip of an Airbus A320 airliner.

**Wingtip sails** Wingtip sails are a form of wingtip device, which consists of usually two or three small lifting surface protruding from the wingtip. Professor Kohn Spillman at Cranfield Institute of Technology in England invented them.

**Wingtip shapes** The shapes of the wingtips varies between different types of aircraft, but the most common shapes past and present are as follows:

- **End-plates**
- **Drooped wingtips**
- **Hoerner tips**
- **Raked wingtips**
- rounded-off
- squared tips
- tip tanks
- **Winglets**
- **Wingtip sails or Tipsail**
- **Wingtip fences.**

*See individual entries.*

**Wingtip stalling** Stalling of the wingtips prior to the root stalling is a problem for some wings, especially sweptback wings. Wingtip stalling has an adverse effect on lateral instability. Although it can be prevented, or at least reduced by using wing wash-out, or fences, etc.

**Wingtip vortices** Wingtip vortices are a product of lift due to the difference in the air pressure gradient above and below the wing, which causes upward recirculation airflow to leak around the wingtips to the low pressure, upper (suction) side of the wing. This action causes wingtip vortices to be generated at all times during flight and accounts for a 5–10% loss in lift at the wingtips, and is the cause of induced vortex drag. The up-flow occurring outboard of the wingtips has little influence on the general airflow over the rest of the wing. However, the down-flow behind the trailing edge intensifies the downwash aft of the wing and hence increases induced drag; it is considered to act through the aerodynamic centre. The combined horizontal free stream airflow

and the induced vertical velocity acting over the wing produce a downwash angle (denoted by  $\epsilon$  epsilon).

Due to the vortex strength being proportional to the wing span, high aspect ratio wings produce weaker vortices and therefore, less induced drag and greater wing efficiency. The relatively narrow chord of high aspect ratio wings, gives the vortices less chance to form, making them weaker. The vortices are also affected by the spanwise flow towards the tips, which can be stronger with sweptback wings. This spanwise flow reduces further inboard towards the wing root, reducing the power of the downwash vortex flow. The vortices are also proportional to aircraft weight and angle of attack, which increases the strength of the vortices – a hazard to following lighter aircraft in the form of wake turbulence.

The vortices flowing up and over the tips of the ailerons can also reduce their effectiveness and degrade lateral stability. Vortices can be reduced by the presence of wingtip fuel tanks, wash-out, fences and winglets, which modify the airflow around the wingtips, reduce the vortex and induced drag.

During suitable adiabatic atmospheric conditions, the high rotational speed of the vortices produce a very low pressure area in the centre causing a temperature drop and if it drops below the dew point a visible vapor trail is formed, especially in high-lift manoeuvres or when approaching to land with full flaps deployed. They are also audible in quiet surroundings, more so after a landing aircraft has just passed the observer.

Vortex dissipation is caused by the two wingtip vortices mixing together some distance behind the aircraft, and general mixing with the atmosphere disturbed by the aircraft's passage. The trailing vortices descend at 300–500 feet per minute to about 1000 feet below the aircraft's cruise altitude before levelling off. This occurs where the air density within the vortices and the ambient density become equal.

Wing tip vortices were first studied in 1907 by Frederick Lanchester (1868–1946), and later by Ludwig Prandtl (1875–1953)

and Theodore von Karman (1881–1963). Further research has been carried out in more recent times.

They are also known as trailing vortices, streamwise vortices, wingtip vortices, and aircraft wake turbulence.

*See* **Winglets; Fence; Aspect ratio; Wash-out.**

**Wing twist** Some modern wings have different airfoil sections built in along the span to take advantage of their different qualities; this is known as aerodynamic twist. However, a wing with the same airfoil section along its whole length, but with the angle of incidence decreasing by two or three degrees towards the tip, has geometric twist – it changes geometrically.

The purpose of wing twist is to control stall behaviour of the aircraft by inducing the wing roots to stall first before the wingtips. Jet transport aircraft with swept wings have very prominent wing twist.

*See* **Wash-out; Aerodynamic twist.**

**Wing warping** Prior to the invention of ailerons, torsional twisting, or wing warping, as it was known, was a rudimentary form of lateral roll control on early aircraft.

The inboard section of the wing is fixed in position while the outer wing panels are allowed to move, or warp, in opposition to provide roll control. This is achieved by lowering the trailing edge to increase the wing camber, angle of incidence and hence lift. On the opposite wing, the trailing edge is raised to decrease the camber and angle of incidence and reduce the lift. The result is the aircraft rolls to induce a turn or to restore the lateral balance after an upset. Wing warping is known as primary warping on the lower wing of a biplane and as secondary warping on the upper wing.

Clement Ader (1841–1925) a French aviation pioneer patented an early form of wing warping roll control in 1890. The ‘Eole’ was the first man-carrying airplane, in which Ader claimed to have

made the first powered flight covering a distance of 164 feet on 9 October 1890; the flight was not officially recognized.

The Wright brothers successfully used wing warping on a five-foot model glider in 1899 and used it on various aircraft they built in the following years, including their now famous Wright Flyer I in 1903. They patented their own type of wing warping for roll control in 1906. The advantage of ailerons, due to their greater effectiveness especially in turbulence, was soon realized and became the common method of roll control for all aircraft that followed. Ailerons were invented in 1908 by Henri Farman (1874–1958).



**Photo:** The Wright brothers used wing warping for lateral control on their early aircraft. The replica Wright Flyer I is displayed at the Pima Air & Space Museum, Tucson, Arizona.

**Work** The definition of work to most people usually refers to a physical activity or employment. However, to the physicist the definition of work involves the mechanical transfer of energy by moving a body a certain distance using a push or pull force. Work done is measured by the product of force times distance moved in the direction of the force. Shown by:

$$\text{Work} = \text{force times distance (} w = Fd \text{)}.$$

Work can also involve transferring energy from one form to another by using a heat process; for example, a piston-engine that transfers chemical energy (fuel) to mechanical energy (thrust). It should be remembered that work could be done either on an object (aircraft) or by an object (engine). The change in kinetic energy is equal to the work done on the object. This can be shown using the SI units of Joules:

$$\begin{aligned}\text{Force} &= 300 \text{ Newtons} \\ \text{Distance} &= 6 \text{ meters}\end{aligned}$$

$$\begin{aligned}\text{Work} &= \text{force times distance (} W = Fd \text{)} \\ &= 300 \text{ N times } 6 \text{ m (} 300 \text{ N} \times 6 \text{ m)} \\ &= 1800 \text{ Nm or Joules of work done.}\end{aligned}$$

Using Imperial units of foot-lb. and given:

$$\begin{aligned}\text{Force} &= 20 \text{ pounds} \\ \text{Distance} &= 3 \text{ feet}\end{aligned}$$

$$\begin{aligned}\text{Work} &= \text{force times distance (} W = Fd \text{)} \\ &= 20 \text{ pounds times } 3 \text{ feet (} 20 \text{ lbs} \times 3 \text{ ft)} \\ &= 60 \text{ foot-lb. of work done.}\end{aligned}$$

- The word Joules is always spelt with a capital J and rhymes with jewels.
- The units are multiplied, not divided.
- The units for work are (N-m) given the name Joule; The Imperial units of foot-lbs has no name.

**Wright brothers** It is common knowledge the Wright brothers, Wilbur (1867–1912) and Orville (1871–1948) are recognized as the first to achieve sustained, powered flight in 1903. However, they were initially frustrated at their first attempts to fly; the wings of their craft were just not performing as expected. Therefore, it was back to the drawing board to find the cure. During 1901–1902, they carried out research on over 200 airfoil sections in their own wind tunnel, which they had built in 1901. In the process, they

recorded the movement of the centre of pressure with changing angle of attack, as a percentage distance from the wing's leading edge. They also corrected the lift formula known as the Smeaton coefficient developed by John Smeaton (1724–1792) an English civil engineer. Smeaton gave the coefficient a value of 0.005, which the Wright Brothers later corrected to 0.0033. [It has since been revised to the lift formula we know today].

Their findings gave them the success they were looking for on 17<sup>th</sup> December 1903. They also designed and built their own propellers for the Wright Flyer.

*See* **Centre of pressure; Smeaton coefficient.**

**Wright-Patterson AFB** Wright-Patterson Air Force Base was also a USAF research and development facility located at Dayton, Ohio, USA, at what was then the Mount Cook facility.

Research into propeller performance was once a major project there when investigating the effects of high Mach numbers, the loss of thrust and increase in drag on World War 1 fighter aircraft propellers.

Two US Army engineers, Frank W. Caldwell and Elisha N. Fales, performed the research. They designed and built their first high-speed 14-inch diameter wind tunnel for airflow-speeds up to 400 Knots (460 MPH). In 1918, a new tunnel was installed for increased flow-speeds approaching Mach 1.0. They also discovered a substantial decrease in the lift coefficient combined with an increase in the drag coefficient with increasing speeds and the first realization of the effects of compressibility. They coined the term 'critical speed', which later became known as the critical Mach number.

This Air Force Base is also home to the National Museum of the United States Air Force, where some of the photographs appearing in this book were obtained by the author.

*See* **Critical Mach number; Drag divergence curve; Lift coefficient & compressibility.**



# X

**X-wing** An X-wing aircraft is a rotary wing helicopter, which is able to continue flight with the rotor blades stationary at 45 degrees to the airflow to provide lift as on a fixed-wing aircraft.

Sikorski Helicopters flight-tested the X-wing concept using a Sikorski S-80 RSRA (rotor system research aircraft) with a first flight of the two helicopters used on 12 October 1976. The S-80 had a modified rotor disc, short wings, and auxiliary jet engines. The cruise speed varied from 140 Knots with the rotor operating as normal, increasing up to a maximum speed of 300 Knots with the rotor stopped.

The thought of a helicopter flying with stationary rotors sounds impracticable. However, the X-wing concept does make it possible with the rotor blades held stationary at 45° to the airflow. The success lies in the use of compressed air from the jet engines, which is ejected through small holes in the rotor blades in the same manner as used in blown flaps. This causes circulation and lift on the rotor blades. Rotating rotor blades always limit the speed of a helicopter in flight; the X-wing concept enabled the helicopter to double its forward speed.

Although the X-wing concept proved it could operate successfully, there are no X-wing helicopters in production to this author's knowledge.

The NASA Dryden Research Centre 1984 completed further flight-testing.

**X-planes** X-planes were so designated by the US Air Force for their research aircraft such as the Bell X-1, Bell X-2 Starbaster, Douglas X-3 Stiletto, Northrop X-4 Bantam, North American X-15 and many more. The US Navy research aircraft were not designated as X-planes.

See **Research aircraft**.

# Y

**Yaw angle** An aircraft's yaw is a movement about the aircraft's OZ vertical axis, either left or right. Yaw, by convention is positive when the aircraft yaws to the right. The yaw angle is the angle between the aircraft's OX longitudinal axis and the reference azimuth (original flight path). The yaw angle is denoted by the symbol  $\psi$  (psi).

**Yaw axis** The aircraft's OZ yaw axis is located at right angles to the OX longitudinal axis and the OY lateral axis. The yaw axis is also known as the OZ normal, vertical axis or side force.

*See Axes.*

**Yaw moment coefficient  $C_n$**  A restoring yaw moment (N) is a force produced by the tail fin/rudder area to rotate the aircraft about the OZ normal axis, after a disturbance from its original flight attitude. A yaw to the right, by convention is considered a positive movement. The most important dimensions of the vertical fin and wing's characteristics, which determine the yaw moment coefficient (N), are the wing area and its span.

The yaw moment coefficient reduces with increasing angle of attack due to the vertical tail/fin being partly immersed in the fuselage boundary layer flow. Sweptback delta wing aircraft are more prone to reduced directional stability (reduced yaw moment coefficient) when operating at high angles of attack, for example when taking-off and landing.

Yaw moment coefficient decreases with increasing Mach numbers due to the vertical fin becoming less effective. A large fin area is required on supersonic aircraft to maintain the required directional stability at high speeds. A dorsal fin mounted atop the fuselage adds vertical fin area and therefore increases the yaw moment coefficient. The yaw moment coefficient and thus

longitudinal static stability is also increased with a rudder-fixed compared to a rudder-free situation.

The minimum control speed ( $V_{MC}$ ) can be calculated for asymmetric flight by graph plotting the yaw moment coefficient from maximum rudder deflection (a straight line) and the yaw moment coefficient from asymmetrical thrust (a curved line). The intersection of the two lines determines the minimum control speed for the aircraft.

The yawing moment coefficient is shown by the formula as follows:

$$N = C_n q S b$$

Where  $N$  = yawing moment

$C_n$  = yaw moment coefficient

$q$  = dynamic pressure

$S$  = wing area

$b$  = wingspan.

**Youngman flaps** The Youngman flap was used on some of the early Fairey Aviation's naval aircraft types. The flaps were attached to struts below the wing's trailing edge to form a slot to improve the airflow over them and thus, make them more effective, similar to Junkers flaps. Apart from their normal use as flaps for take-off or landing, they also had the ability to deflect upwards by about thirty degrees to act as a dive brake with little effect on trim changes. They were used in WWII on the Fairey Swordfish, Albacore, Firefly and the Barracuda torpedo dive-bombers, for example.

They were designed by Robert Talbot Youngman and patented by the Fairey Aviation Company Ltd, England. They are also known as Fairey-Youngman flaps.



**Photo:** The Fairey Firefly was one aircraft to be equipped with Youngman flaps. They are shown in this photograph retracted on the folded wings. The aircraft is on display in the Fleet Air Arm Museum in Yeovil, England.

# Z

**Zap flap** The zap flap is a combination of the slotted flap and the split flap. It travels rearwards when deployed and then rotates downward. It did not prove very popular with aircraft designers; one exception is the Northrop P-61 Black Widow night-fighter. The German, Edward F. Zaparka invented the Zap flap in 1932.

**Zero lift** The wing's angle of attack where no lift is produced is known as zero lift.

*See Zero lift line.*

**Zero lift axis** *See Zero lift line.*

**Zero lift drag  $C_{D0}$**  Zero lift drag is an alternate name for the aircraft's parasite drag. It includes all non lift-dependant drag such as the profile drag, skin friction drag, form drag, body and extra drag, plus rotor blade drag on a helicopter. It increases by the square of the velocity increase – double the speed and zero lift drag increases four fold.

Zero lift drag is also known as parasite drag.

**Zero lift line** The zero lift line relates to an angle of attack at which an airfoil produces zero lift.

On a cambered airfoil, the zero lift limit is reached when its chord line reduces to about minus two or three degrees angle of attack and it is moving parallel to the zero lift line. When a symmetrical airfoil's chord line is at zero degrees angle of attack, which is inline with the chord line, it is moving parallel to its zero lift line and hence no lift is developed.

The propeller's zero lift line is related to a certain negative angle of attack (and experimental pitch) at which the cambered airfoil produces zero lift and therefore, zero thrust. **Diagram 68,**

**Propeller Pitch**, which shows the vector A-E zero thrust line with the cambered propeller blade operating at its experimental pitch and producing zero thrust. From this diagram, the reader can imagine a wing's cambered airfoil operating at a negative angle of attack (vectors A-E and A-D) will also be producing zero lift.

It can also be stated; when the zero lift line is parallel to the relative airflow the wing, produces zero lift (or in the case of the propeller, zero thrust).

Also known as the no-lift angle, line of zero lift or zero lift axis.

See **Diagram 49, Lift Coefficient V. Angle of Attack.**

**Zero lift pitching moment ( $C_{m0}$ )** On an aircraft's cambered wings operating at zero degree angle of attack a small amount of lift is still being produced. A further reduction in angle of attack to a negative value will cause a pitching moment to become established when the centre of pressure above and below the wing produce a turning or zero lift pitching moment. This will cause the aircraft's nose to pitch further downwards.

See **Zero lift line.**

**Zero net thrust** The propeller produces zero net thrust when operating at a negative angle of attack (experimental pitch) which is equal to the advance per rev. This is shown on **Diagram 68, Propeller Pitch**, as the zero thrust line vector A-E.

See **Experimental pitch.**

**Zero rate of climb speed** The delta wing does not stall in the normal sense of the word and so it does not have a specific stall speed. Therefore, the speed for the zero rate of climb is used as a reference speed in lieu of the stall speed.

See **Delta wing.**

**Zero reference plane** The zero reference plane commonly starts at the tip of the aircraft's nose, or any designated station in front of or behind the nose, in conjunction with the start of the zero reference line. The reference plane (which is normal to the fuselage axis) and the zero reference line are used for aerodynamic geometries such as centre of gravity and load calculations.

**Zero torque pitch** The zero torque pitch is the propeller blade's angle of attack when the propeller is windmilling with zero torque.

*See Windmilling propeller forces; Mach angle & cone.*

**Zhukovski, N. Prof** Professor Nikolai Zhukovski (1847–1921) the 'father of Russian aviation' was also known as Professor Nikolai Yegerovitch Joukowski. The name Zhukovski is the English transliteration spelling for his Russian name.

*See Joukowski, N. Prof.*

**Zone of silence** The undisturbed free air stream outside of the Mach cone attached to a supersonic plane is known as the zone of silence. The term also applies when shock diamonds appear in a supersonic jet exhaust flow. The space between the exhaust nozzle and the first shock diamond is known as the zone of silence.

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